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LIQUID ROCKET SPACE ENGINES

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Liquid rocket engines (ZhRD), which operate on liquid fuel, along with solid fuel rocket engines (RDTT) are the main types of space engines. They are used extensively in cosmonautics. Man's daring dream -- travel into space -- was accomplished by using liquid rocket engines. On 4 October 1957 the world's first launch of an artificial earth satellite, which opened the space age in mankind's history, occurred in the USSR using powerful liquid rocket engines and on 12 October 1961 liquid rocket space engines provided orbital insertion of a spacecraft around the earth with the first cosmonaut -- Yu. A. Gagarin.

Almost all space rocket launches are now accomplished using liquid rocket engines. Liquid rocket engines are used extensively and directly in space during spacecraft flights: artificial earth satellites, automatic interplanetary stations, spacecraft* and so on.

The numerous liquid rocket space engines may be divided into three groups: the main engines of space rockets, the main engines of spacecraft and auxiliary engines. Main liquid rocket engines are used in space operations which require high specific expenditures: during acceleration of space rockets, during acceleration and braking of spacecraft, transfer of spacecraft to other orbits and so on. Auxiliary engines support operations unrelated to high energy expenditures: staging and space rocket flight control, spacecraft orientation and stabilization and so on. It should be said that main liquid rocket engines also frequently fulfill auxiliary functions. Moreover, clear delineation of liquid rocket engines into main and auxiliary is not always possible.

General Data on the Operating Principle of Liquid Rocket Space Engines

Liquid rocket engines operate on the principle of conversion of the potential chemical energy of liquid fuel to the kinetic energy of gases flowing from the engine; in this case an emissive force occurs, directed opposite to the gas flow and called reactive force or thrust. It should be noted that a liquid rocket engine is only part of a rocket power plant which also contains fuel tanks, fittings and pipelines which connect the liquid rocket engine to the tanks.

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*Space rockets are also frequently related to spacecraft in scientific and technical literature. In this pamphlet we shall call spacecraft those enumerated objects which are space rocket payloads.

Liquid rocket engine fuel may be bipropellant and monopropellant. Bipropellant consists of a liquid oxidizer (oxygen, nitrous oxides and so on) and a liquid fuel (hydrogen, hydrocarbons and so on) stored in separate tanks. Mon propellant is a liquid (for example, hydrazine) capable of catalytic decomposition.

Conversion of liquid fuel to a reactive gas jet occurs in a chamber which is the main and indispensable component of any liquid rocket engine. The chamber operating on bipropellant contains a combustion chamber in which the oxidizer and the fuel interact with each other (burn) with formation of a high-temperature gas, and a propelling supersonic nozzle in which the formed gas is accelerated to a velocity exceeding the speed of sound. Total fuel combustion is achieved by a preliminary atomizer and by mixing of the oxidizer and fuel using a mixing head equipped with injectors. The gas temperature in the chamber reaches several thousand degrees and continuous cooling of it is therefore required to maintain the integrity of the chamber under these conditions. It may be accomplished, for example, by using fuel flowing through channels in the chamber housing prior to entering the mixing head. This method of oxidation is called regenerative.

A chamber operating on a bipropellant and its working process are shown in Figure 1.

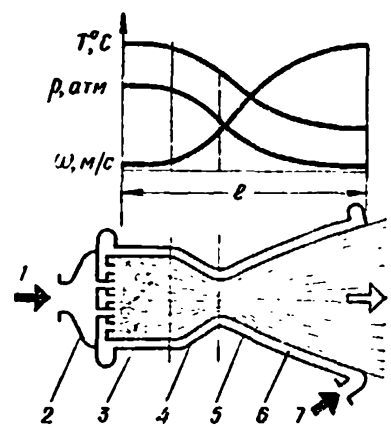

Figure 1. Diagram of Liquid Rocket Engine Chamber Operating on Bipropellant and Variation of Gas Parameters Along Its Length l: 1 -- oxidizer; 2 -- mixing head; 3 -- combustion chamber; 4 -- subsonic part of nozzle; 5 -- supersonic part of nozzle; 6 -- cooling channel; 7 -- fuel

A liquid rocket engine chamber operating on monopropellant contains a so-called expansion chamber in which the liquid fuel is converted to a gas in the presence of a catalyst. It may either by the liquid entering the chamber from a separate tank or a solid located in the chamber itself.

Liquid rocket space engines are divided into pressure fuel feed and with pump feed engines. The fuel in liquid rocket engines of the first type enters the chamber from tanks due to being forced out by gases entering, for example, from special tanks. A liquid rocket engine with pump feed is distinguished from that described by the presence of a turbopump unit and gas generator. The turbopump unit contains fuel pumps (usually of the centrifugal type) and a gas turbine which sets them in motion, rotated by a gas which is produced in the mentioned gas generator; spent gas may be exhausted to an exhaust pipe.

A liquid rocket engine with pump fuel feed is shown in Figure 2 in the composition of an engine power plant.

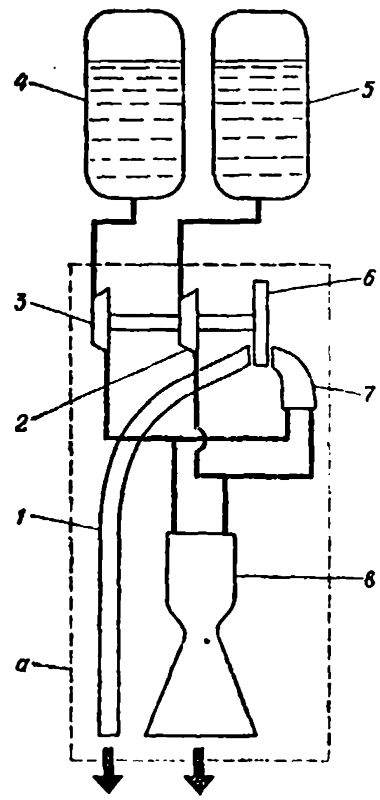

Figure 2. Diagram of Engine Power Plant With Liquid Rocket Engine Having Pump Feed of Fuel: a -- liquid rocket engine; 1 -- turbine exhaust pipe; 2 -- oxidizer pump; 3 -- fuel pump; 4 -- fuel tank; 5 -- oxidizer tank; 6 -- turbine; 7 -- gas generator; 8 -- chamber

In the general case the thrust of a liquid rocket space engine consists of the chamber thrust (the main component) and the thrust of the turbine exhaust pipes. The chamber thrust is determined in this case as $P_k = m\omega_a + F_a(p_a - p_n)$, where m is the fuel mass expended by the chamber within 1 second; w_a is the gas flow velocity; F_a is the output cross-sectional area of the chamber; p_a is the gas pressure at the chamber output; and p_n is the pressure of the surrounding medium.

The ratio of thrust to fuel consumption (denoted as I_u) has the dimension of velocity and is called specific impulse (of a liquid rocket engine or the chamber). This parameter characterizes liquid rocket engine efficiency. It obviously follows from the well-known Tsiolkovskiy formula, which may be written as $\ln M_0/M_k = v_k/I_u$ (here M_k and M_0 are the final and initial masses, respectively, of a rocket unit and v_k is the final velocity of the apparatus), but even a slight increase of I_u leads to a significant increase of the permissible ratio M_k/M_0, i.e., all things being equal, to a significant increase of the rocket payload. As can be seen from the given formula for P_k, the thrust and specific impulse of a liquid rocket engine increase as flight altitude increases. Since the first term in this formula is considerably greater than the second and, moreover, since the main fuel mass of the liquid rocket engine flows through the chamber, the specific impulse of a liquid rocket engine is approximately equal to the gas flow rate from the chamber and may be calculated in the following manner: $I_u \simeq w_a = 129\sqrt{T_k[1 - (p_a/p_k)^{\lambda}]/\mu\lambda}$, where T_k is the gas temperature in the combustion chamber (expansion); p_k and p_a are the gas pressure in the combustion chamber and at the nozzle output, respectively; μ is the molecular weight of the gas in the combustion chamber; and λ is a coefficient which characterizes the thermophysical properties of the gas in the chamber (usually $\lambda \simeq 15$).

The values of parameters t_k, μ and λ are mainly determined by the type of rocket fuel. If the specific heat of the fuels is identical, a greater value of T_k and consequently of I_u corresponds to products of greater calorific value. Parameter μ determines the gas productivity of the fuel: the smaller μ, the more gases are formed in the chamber and the higher is I_u.

Much higher values of I_u correspond to bipropellants than to monopropellants. Unfortunately the fuels which provide the highest specific impulse have low density and consequently require cumbersome (and heavy) fuel tanks for their disposition, which reduces the efficiency of these fuels.

If parameters T_k, μ and λ are determined by the type of rocket fuel, the values of p_k and p_a are selected during engine design. Liquid rocket engines which launch space rockets are usually designed for p_a = 0.4-0.7 atm and those operating during the last stages of space flight are usually designed for smaller values of p_a (up to hundredths of an atmosphere). When selecting the value of p_a, the fact that the overall dimensions and weight of the nozzle increase as this parameter decreases is particularly taken into account.

An increase of pressure in the combustion chamber (p_k) is a real means of increasing the specific impulse since the ratio p_a/p_k decreases as pressure p_k increases (for a selected value of p_a).

An increase of pressure p_k also leads to a decrease of the overall dimensions of the chamber. For these reasons the pressure in the chamber is related to the number of main parameters which characterize the degree of technical perfection of liquid rocket engines. The value of p_k for liquid

rocket space engines with pressure fuel feed usually comprises approximately 10 atm; the value of p_K for liquid rocket engines with pump feed is considerably higher and reaches hundreds of atmospheres. The latter engines are better than those of the first type both in economy and overall dimensions and in weight ratio: they are characterized by a small specific weight (the weight of liquid rocket engines in kilograms per 1 ton of thrust).

Although the fuel tanks in the case of a liquid rocket engine with pump feed are under gas pressure (tank boost pressure), this pressure is less than in the case of a liquid rocket engine with pressure feed and consequently the fuel tanks are lighter.

Pump feed is now used only in liquid rocket engines operating on bipropellants. With rare exceptions, all liquid rocket engines with thrust greater than 10 tons contain turbopump assemblies.

Let us note in conclusion that the following are taken into account when selecting the type of rocket fuel, operating parameters and layout of liquid rocket engines (along with the general concepts outlines): the possibility of storing the rocket fuel under space flight conditions, the operating mode of the liquid rocket engine, the reliability of the rocket engine power plant and many other factors.

The History of Development and Use of Liquid Rocket Engines

The liquid rocket engine was first proposed by our own K. E. Tsiolkovskiy in 1903 as an engine for space flight. Tsiolkovskiy defined development of powerful, economic liquid rocket engines as a primary problem on the path toward space flight.

Practical investigations on development of liquid rocket engines were begun in 1921 by the American R. Goddard, who somewhat later in 1926 launched a small rocket with a liquid rocket engine. Development of liquid rocket engines was begun in the USSR, Germany and other countries at the end of the 1920's and beginning of the 1930's. The first Soviet experimental liquid rocket engine, the ORM-1, designed by V. P. Glushko and developed at the Gas Dynamics Laboratory (GDL), was tested in 1931.

Experimental models of liquid rocket engines with thrust up to several hundred kilograms, designed for experimental flying vehicles, appeared in the USSR and the United States prior to the beginning of World War II. Serially-produced liquid rocket engines were developed in a number of countries by the end of the war. The first Soviet serially-produced liquid rocket engines were those of the RD-1 type with a thrust of several hundred kilograms, designed for aircraft. They were developed at the experimental design office, which was subsequently known as GDL-OKB.

An important engineering achievement was development of the first large liquid rocket engines developing a thrust of more than 25 tons in the 1940's. Ballistic missiles with a range of several hundred kilometers and also

geophysical rockets which lifted scientific apparatus to high altitudes were developed on the basis of these liquid rocket engines.

By the middle of the 1950's liquid rocket engines were subjected to a number of improvements and the range of the rockets exceeded 1,000 km. The capabilities of the type of liquid rocket engine developed by that time were essentially exhausted in this case due to the low specific heat of the rocket fuel (oxygen-ethyl alcohol) used at that time, the inefficiency of its use and also the imperfection of the engine design of this type

The time that ballistic missiles designed for a range of several thousand kilometers and of space rockets, development of which was begun in the USSR and the United States in 1954-1955, appeared was determined largely by the capability of developing a new type of liquid rocket engine which far exceeded existing engines in all main characteristics. Investigations to develop new types of liquid rocket engines, begun in the USSR and the United States even during the second half of the 1940's and which were conducted parallel with improvement of existing engines, became one of the main prerequisites for development of space rockets.

The Soviet space rocket which was used to launch the world's first artificial satellite in 1957 was equipped with a liquid rocket engine developed at GDL-OKB under the supervision of the prominent Soviet scientist V. P. Glushko.

Main Liquid Rocket Space Engines

As noted above, liquid rocket engines are the main types of engines which are used to accelerate most space rockets. The number of all models of liquid rocket engines used for this purpose is many thousands. Most Soviet liquid rocket engines were developed at GDL-OKB; they were installed on all Soviet rockets launched into space. Development of liquid rocket engines for space rockets abroad is related primarily to the American Rocketdyne Company and to the work of a group of engineers supervised by S. Hoffman. The liquid rocket engines of this company were used to boost more than 80 percent of the American space rockets launched since 1958.

Development of space rocket engines is related to enormous expenditures of funds. For this reason only four countries except the United States and the USSR have been able (1965-1971) to launch space objects (artificial earth satellites) through their own efforts. These countries include France, Japan, the Chinese People's Republic and Great Britain (enumerated in the chronology of the first launches).

Having launched only a single artificial earth satellite through their own efforts, Great Britain, like a number of other countries, now enjoys the use of United States services to launch space apparatus. Japan, being limited to several launches of low-power space rockets, has now begun to produce an American rocket under license, on one stage of which a liquid rocket engine of Japanese design developing a thrust of several tons has

been mounted. France continued to carry out infrequent launches of space apparatus by using a low-power rocket with a liquid rocket engine on the first stage. Some West European countries (France, West Germany and others) are developing the European "Ariane" space rocket with liquid rocket engine through their combined efforts. India is developing a space rocket with RDTT from a model of the American "Scout" rocket. Launch of satellites using the indicated rockets is planned at the end of the 1970's and beginning of the 1980's.

We shall subsequently devote our attention primarily to powerful liquid rocket space engines with a thrust of several tens and hundreds of tons. Therefore, the main position in the pamphlet will be devoted to engines of the USSR and the United States since liquid rocket space engines in these countries, like cosmonautics as a whole, have been developed extensively. Let us begin the story of these liquid rocket engines with the world's first space rocket engines.

The First Space Rocket Engines (History of Development, Design and Parameters)

The liquid rocket engines designed by GDL-OKB for the first space rocket were the result of extensive scientific research and experimental design work begun in the USSR 10 years before the flight of the first space rocket. New designs, fuels, materials, production processes and equipment were developed and assimilated as a result of these investigations. The main technical solutions utilized in the first space engines were primarily checked at GDL-OKB during preliminary development of several types of liquid rocket engines brought to different stages of accomplishment. The first of them was an engine with a thrust of more than 120 tons, designed to use oxygen-kerosene bipropellant. Development of this liquid rocket engine during the period 1947-1951 reached the stage of manufacture and testing of various subassemblies.

The primary result of all these investigations was development of the liquid rocket engine chamber, capable of operating at high temperatures and pressures. The chamber design used in the large liquid rocket engines of the 1940's clearly did not meet this requirement. The chamber walls had to be made thick from strength concepts and the chamber was massive and heavy. Strength requirements very soon entered an unresolvable contradiction with cooling requirements upon augmentation of the chamber operation.

A chamber of essentially new design (Figure 3) was suggested, developed and introduced into Soviet technology. The body of this chamber, designed for regenerative cooling, consisted of inside and outside walls (called the fire wall and jacket, respectively), joined to each other by soldering through the intermediate corrugated wall (interlining) or through ribs cut in the fire wall. The chamber contained a mixing head with injectors soldered into it and was divided technologically into several subassemblies joined in the final phase of the production process by annular weld seams.

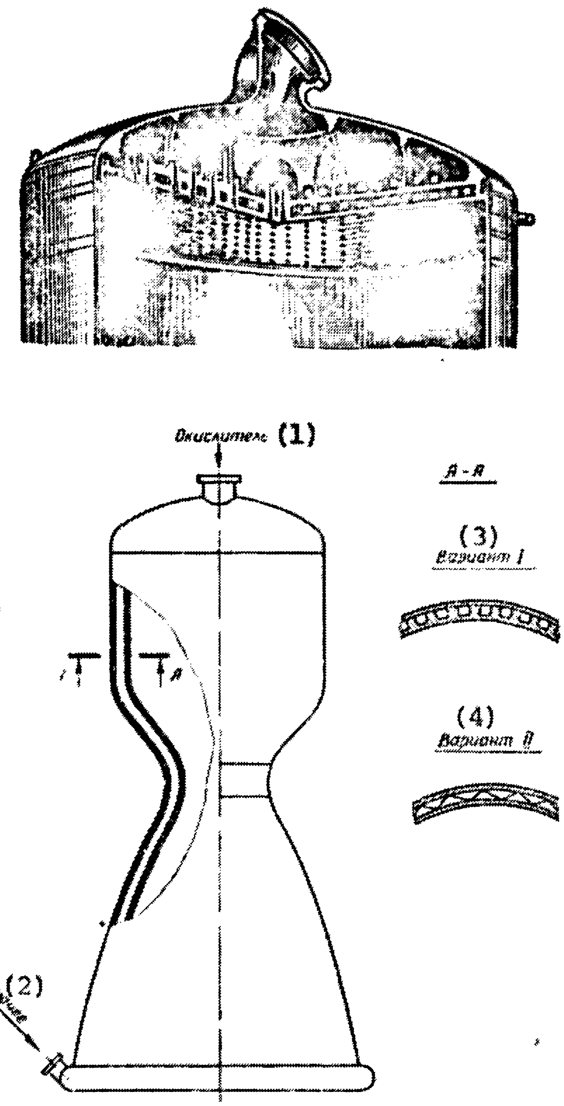

Figure 3. Design of Soldered-Welded Chamber (the Mixing Head of the Chamber Is Shown Above in Cross-Section)

KEY:

1. Oxidizer
2. Fuel
3. Variant I
4. Variant II

Due to the presence of special couplings in the body of this chamber, its fire wall can be made thin and can be manufactured from relatively weak, but high heat-conducting copper alloys; reliable cooling of the chamber is thus provided. The jacket which receives gas pressure was manufactured from alloy steels providing high strength of the chamber with low weight. The capability of significantly increasing the economy of liquid rocket engines appeared with development of soldered-welded chambers (due to increasing the combustion chamber pressure and using efficient rocket fuels) with a considerable reduction of the specific weight of the engine.

Numerous test-firings of small experimental chambers preceded manufacture of full-scale soldered-welded chambers. As a result the efficiency of the new type of chamber was checked extensively and the correctness of selection of oxygen and kerosene as the fuel was confirmed. This fuel pair exceeded the oxygen-ethyl alcohol fuel pair then used in liquid rocket engines in both specific impulse and in density and provided the longest flight range of the rockets. The soldered-welded chambers were designed for a pressure two-four times greater than that at which oxygen-alcohol liquid rocket engines operated. The use of kerosene and an increase of pressure resulted in an increase of temperature by 800°C in the combustion chamber. Subsequent development of liquid rocket engines showed, however, that the soldered-welded chamber is also efficient under much more severe conditions.

Development of soldered-welded chambers largely determined the success of developing the engines which provided the flight of the first space rocket, which is a two-stage vehicle consisting of a central pod (second stage) and four lateral pods (the first stage). An RD-108 liquid rocket engine was installed in the central pod and an RD-107 was installed in the lateral pods (RD is an abbreviation of the name "rocket engine"). All liquid rocket engines were fired simultaneously, providing a launch thrust of approximately 410 tons. The lateral pods were separated about 120 seconds after launch (flight altitude in this case was approximately 50 km and velocity was 3,200 m/s); the central pod continued to operate for an additional 180 seconds, accelerating the payload to orbital velocity.

The RD-107 and RD-108 engines are similar in characteristics since an attempt was made to standardize them to the maximum. The main engine assemblies are distinguished only by the operating parameters (they are somewhat higher for the RD-107. An overall view and a schematic diagram of one of the liquid rocket engines (the RD-107) are presented in Figures 4 and 5.

The RD-107 and RD-108 develop a thrust of 102 and 96 tons, respectively, which is approximately 90 percent developed in each liquid rocket engine by four identical chambers fed from a single turbopump assembly. A total of 52 kg of oxygen and 21 kg of kerosene enters the RD-107 chamber within 1 second. The oxygen is delivered directly to the mixing head through a central pipe and the kerosene is delivered to an annular collector located near the chamber outlet, from which it is distributed through the cooling channels and then, heated to 210°C, is fed to the mixing head. The fuel is vaporized by 337 injectors, of which one is installed in the center and the remaining ones are located in 10 concentric circles. The kerosene injectors, which create a protective gas-liquid film near the chamber fire wall, are installed near the periphery. The heat flux to the wall reaches a maximum value -- more than 14 million kcal$\cdot$(m^{2}hr)$^{-1}$ -- in the region of minimum chamber cross-section. The cooling chamber of the liquid rocket engine is designed here for maximum coolant flow rate -- up to 20 m/s and the fire wall temperature reaches a maximum value -- 380°C. Upon combustion of fuel, a gas with pressure of 60 atm and temperature of 3250°C forms in the

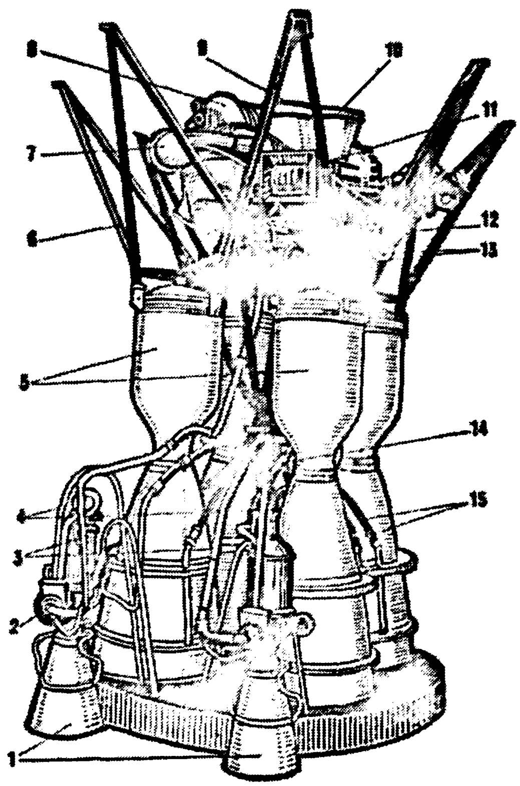

Figure 4. RD-107 Liquid Rocket Engine: 1 -- steering chambers; 2 -- steering chamber rotation assembly; 3 -- pipelines for delivery of oxidizer to steering chambers; 4 -- mock-up brackets (not included in the liquid rocket engine design); 5 -- main chambers; 6 -- frame for attaching liquid rocket engine to rocket; 7 -- gas generator; 8 -- turbine; 9 -- oxidizer pump; 10 -- fuel pump; 11 -- chamber pressure sensor; 12 -- main oxidizer valve; 13 -- pipelines for delivery of oxidizer to chambers; 14 -- main fuel valve; 15 -- pipelines for delivery of fuel to chambers

combustion chamber. The pressure drops to 0.4 atm and temperature drops to 1690°C after the gas passes through the nozzle. The gas is accelerated in this case to a velocity of 2,950 m/s, imparting a thrust of more than 23 tons to the chamber.

Success in developing the RD-107 and RD-108 engines largely determined development of a compact, lightweight turbopump assembly capable of operating at sufficiently low fuel pressure in the rocket tanks (Figure 6). The turbopump assembly of these liquid rocket engines contains two main and two auxiliary pumps and the turbine which rotates them, whose output is 5,200 hp for the RD-107. The pumps are of the centrifugal type and single-stage; the turbine is axial and two-stage. The main pumps are used to feed fuel to the chambers. They are installed coaxially with the turbine and have identical rotational speed with it -- 8,300 rpm.* The pumps are designed to feed

*This value and also those pump and gas generator parameters given later refer to the RD-107 engine.

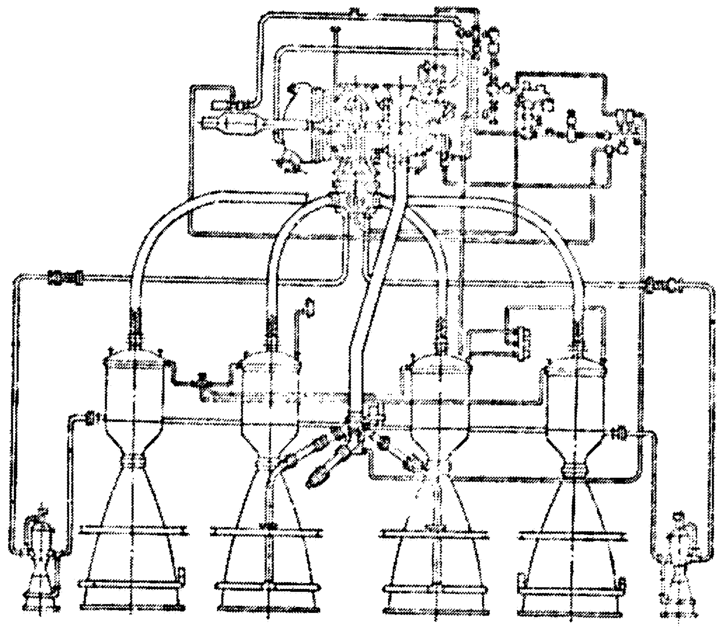

Figure 5. Schematic Diagram of RD-107 Liquid Rocket Engine

oxygen at a rate of 226 kg/s under a pressure of 80 atm and to feed kerosene at a rate of 91 kg/s under pressure of 95 atm. Cavitationless operation of the pumps is provided by installation of low-pressure axial vanes in front of the main rotors.

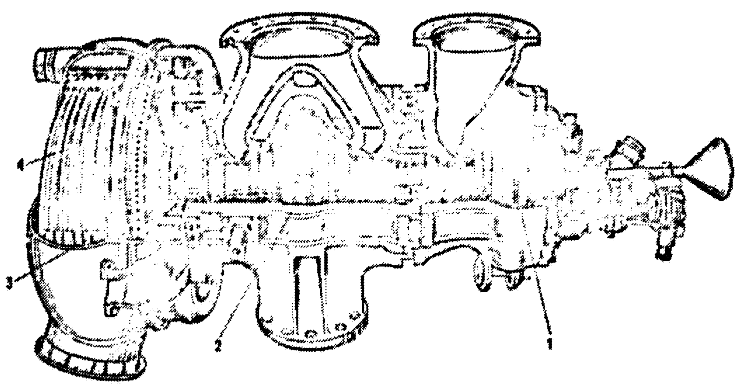

Figure 6. Turbopump Assembly of the RD-107 and RD-108 Engines: 1 -- fuel pump; 2 -- oxidizer pump; 3 -- turbine; 4 -- heat exchanger

The auxiliary pumps are driven by a booster. One of the pumps delivers liquid nitrogen to a vaporizer built into the turbine exhaust collector and used to deliver the gases for supercharging the rocket fuel tanks. The other auxiliary pump delivers low-water (82 percent) hydrogen peroxide to the gas generator, which produces vapor-gas to drive the turbine.

The gas generator of the liquid rocket engine for the first space rocket was a cylindrical tank which contained a granular substance -- a catalyst. Upon passing through it, the hydrogen peroxide was decomposed into a mixture of water vapor and gaseous oxygen with a pressure of 55 atm and temperature of 560°C; this mixture was fed at a rate of approximately 9 kg/s to the turbine blades. The spent gas with a pressure of 1.5 atm and temperature of 200°C was ejected through the exhaust pipes at a velocity of 450 m/s. A thrust of approximately 700 kg was developed in this case.

Development of the RD-107 and RD-108 engines was combined with solution not only of purely "engine" problems but of problems of general improvement of rockets which entrusted a number of functions previously not inherent to them to liquid rocket engines. One of these functions was to provide the rocket flight with a strictly given trajectory and to control the rocket position. Rotary steering chambers equipped with hollow journals through which fuel was fed from the main pumps of the turbopump unit and which provided declination of the chambers at an angle of +45°, accomplished by means of hydraulic drives, were provided in the liquid rocket engine design to fulfill the indicated function. There are two steering chambers in the RD-107 and four in the RD-108 engine. The steering chamber is similar to the main chamber in design. However, operating at almost the same pressure as the main chamber of the RD-107, the steering chamber developed six times less thrust.

Assemblies to control the thrust and the ratio of fuel component flow rates were provided in the engine of the first space rocket, which permitted flight support at a previously calculated optimum velocity and made it possible to achieve total synchronous exhaustion of the fuel components from the rocket tanks.

The oxygen-kerosene fuel used in the RD-107 and RD-108 engines is not self-combustible; it is ignited by means of pyrotechnic devices introduced into the liquid rocket engine chamber from the nozzles and which are triggered when an electric current is supplied. The operation of these liquid rocket engines is controlled by electro-, pneumo- and pyroautomatics.

With regard to all the elements in the set, the weight of the RD-107 engine is equal to 1,155 kg, which corresponds to a specific weight of 11.3 kg/t (the RD-108 is 95 kg heavier). The specific impulse of the RD-107 and RD-108 is 3,080 and 3,090 m/s, respectively. For comparison let us point out that the best models of oxygen-alcohol liquid rocket engines, which preceded the appearance of space engines, were characterized by a specific weight of 17 kg/t and specific impulse of 2,430 m/s.

Brief Summary of Space Rocket Development

A two-stage space rocket with RD-107 and RD-108 engines was used in 1957-1958 to launch the first artificial earth satellites. A third stage with a single-chamber oxygen-kerosene liquid rocket engine having thrust of 5 t, developed at the OKB headed by S. A. Kosberg, was developed for the

indicated rocket to launch automatic interplanetary stations (AMS) to the moon (1959) and the manned orbital flights of the "Vostok" spacecraft (1961). The three-stage rocket, designed in this manner, was named the "Vostok." Later modifications of it with a four-chamber liquid rocket engine having a thrust of 30 t in the third stage has been used since 1964 to launch the "Voskhod" spacecraft and since 1967 to launch the "Soyuz" spacecraft.* AMS, which were inserted into a circumlunar orbit and which made a soft landing on the Moon (1966), Venus (1970) and Mars (1971), were launched by four-stage modifications of the "Vostok" rocket.

"Vostok" carrier rockets provided the USSR for a long time with the leading role in space research and continue to be used extensively up to the present time. Up until 1964 they considerably exceeded in capacity all other space rockets, among which may be named such liquid-fueled rockets as the Soviet "Kosmos" (used since 1962) and the American "Thor-Delta" and "Atlas-Agena" rockets (used since 1960). These American space rockets, like the "Vostok," were developed by installing additional stages on the just developed liquid-fueled ballistic missiles with a range of several thousand kilometers (the first space rocket was a variant of an intercontinental ballistic missile).

More powerful space rockets than "Vostok" rockets were developed in subsequent years in the USSR and the United States. They include the "Proton" (USSR), "Saturn-1" (two versions), "Saturn-5" and the liquid-fueled rockets of the "Titan-3" family with launch RDTT. Space rockets of the "Titan-3" family were developed on the basis of intercontinental ballistic missiles and the remaining ones are completely original designs.

The "Proton" and the latest versions of the "Sature-1" and "Titan-3" rockets have approximately three times the lifting power of the modified models of the "Vostok" rocket. Many outstanding achievements of cosmonautics, which will be discussed further during description of corresponding rocket engines, are related to their use.

The "Saturn-5" space rocket exceeds all other rockets in lifting power. It was developed within the "Apollo" program, whose purpose included delivery and landing of astronauts on the moon. A rocket with a launch mass of approximately 3,000 t was required for this. After six expeditions to the moon (1969-1972), the "Saturn-5" has been used only once -- to launch the manned orbital station "Skylab" (1973) with a mass of approximately 80 t.

The appearance of more powerful space rockets than those of the "Vostok" type became possible only due to remarkable new achievements in the field of liquid rocket engine development. They include development of engines with chambers having thrust of more than 100 t, development of new rocket fuels and achieving pressures exceeding 100 atm in the combustion chambers. We shall consider all these achievements later using specific liquid rocket

*These modifications of the "Vostok" rocket are sometimes called the "Voskhod" and "Soyuz" by analogy with the names of the spacecraft.

engine models as an example and we shall begin our story with description of the F-1 engine which is used on the first stage of the "Saturn-5" rocket.

The F-1 Liquid Rocket Engine

This engine, designed by the Rocketdyne Company, is shown in Figure 7. It operates on oxygen-kerosene fuel and develops a thrust from 690 (during launch) to 790 t (prior to stage separation). The F-1 is a single-chamber liquid rocket engine with pump feed of the fuel. The gas for driving the turbopump assembly is produced in a gas generator by combustion of the main fuel of the liquid rocket engine (i.e., the fuel on which the chamber operates), but with a large excess of fuel (a diagram of this liquid rocket engine is shown in Figure 2). Approximately 3 percent of the total flow rate of fuel through the liquid rocket engine is expended through the gas generator. Gas with a temperature of approximately 800°C, which is ejected into the nozzle section of the chamber after driving the turbine, is formed in the gas generator. The gas generation system of the described type is used extensively in modern liquid rocket engines (the spent turbine gas is frequently injected into the exhaust pipe).

The F-1 exceeds all other foreign liquid rocket engines in chamber pressure, which comprises 70 atm. A so-called tubular chamber whose body is formed by shaped tubes joined to each other by soldering (the tubes form the regenerative cooling channel of the chamber), is used in the F-1 engine. The mixing head of the chamber is detachable with openings which perform the role of oxidizer and fuel atomizer nozzles. Tubular chambers are typical to foreign liquid rocket engines. One of them is shown in Figure 8.

The F-1 engine is fired by "gravity flow" of the fuel: when the starting valves are opened, the oxidizer and fuel are delivered to the chamber and gas generator at tank supercharging pressure in the nonoperating pumps; when the fuel in the gas generator is ignited, the produced gas turns the turbopump assembly. This launch scheme, which does not require auxiliary devices to turn the turbopump assembly, was first accomplished in the Soviet liquid rocket space engine "Proton."

The fuel in the chamber is ignited during launch of the F-1 engine by a starting fuel (a mixture of triethyl aluminim and triethyl borane), distributed in a special sleeve and ignited upon contact with the oxidizer. This method of ignition, called the chemical method, is used extensively in modern liquid rocket space engines (the launch fuel may be located directly in the pipelines of the liquid rocket engine).

The assemblies which control operation of the F-1 are triggered by electric commands and the fuel pressure selected from the high-pressure mains, which eliminates the requirement for auxiliary compressed-gas systems.

Five F-1 engines are installed on the "Saturn-5" rocket: the central engine is fixed and the peripheral engines are in gimbal suspensions which provide rotation of the engine in its own plane parallel to its axis to control the

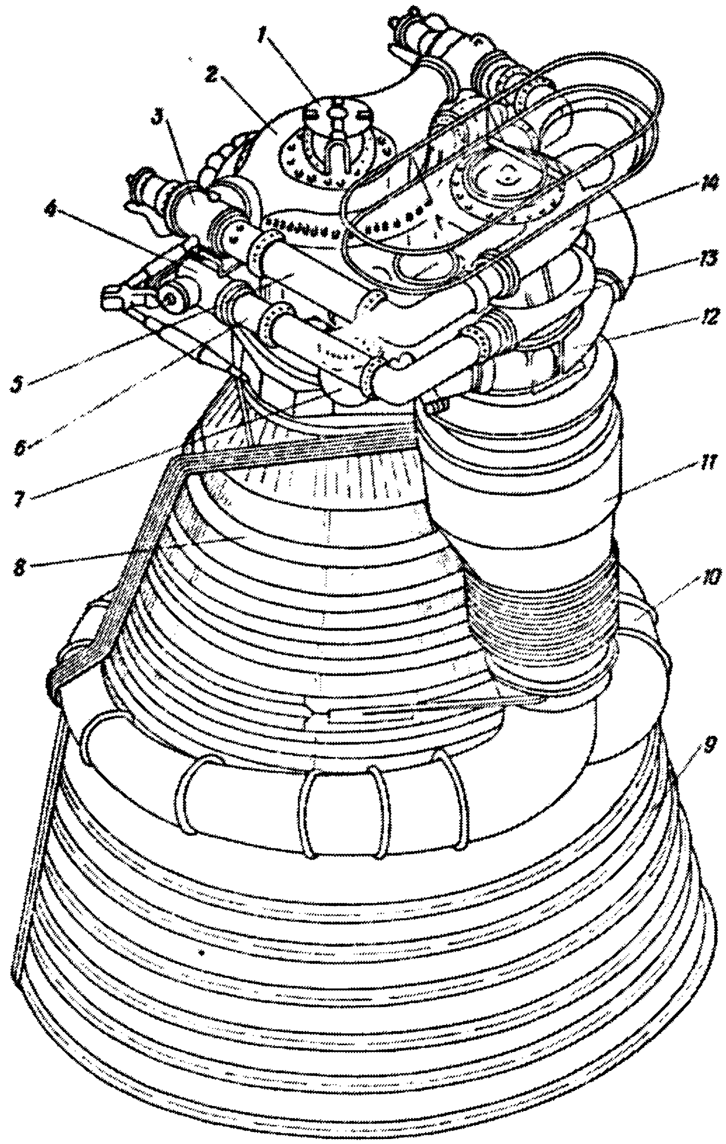

Figure 7. F-1 Liquid Rocket Engine: 1 -- gimbal suspension; 2 -- oxidizer to chamber feed collector; 3 -- main oxidizer valve; 4 -- main fuel valve; 5 -- high-pressure oxidizer pipeline; 6 -- high-pressure fuel pipeline; 7 -- gas generator; 8 -- tubular part of chamber; 9 -- removable part of nozzle; 10 -- assembled collector of spent turbine gas; 11 -- heat exchanger; 12 -- turbine; 13 -- fuel pump; 14 -- oxidizer pump

flight direction and position of the rocket. Let us present some of its parameters for clear representation of the scale of the F-1 engine. The engine mass is 8,400 kg and height is 5.6 m; the chamber diameter in the output section is equal to 3.6 m. The engine consumes approximately 2.5 t of fuel within 1 second, which is delivered by a turbopump assembly having capacity of 55,000 hp.

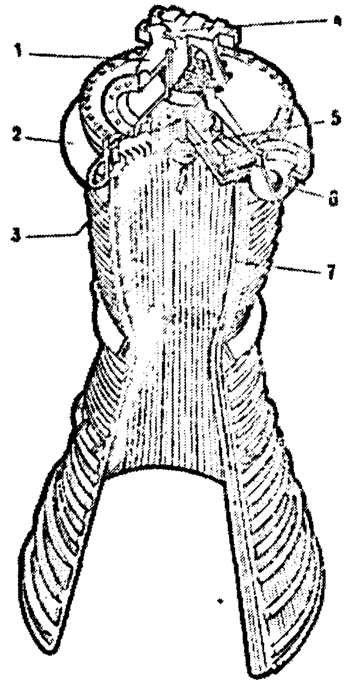

Figure 8. Tubular Chamber of Liquid Rocket Engine: 1 -- oxidizer feed pipe; 2 -- fuel feed collector; 3 -- mixing head; 4 -- gimbal suspension; 5 -- ignition device; 6 -- fuel feed pipe; 7 -- tubular body of chamber

Hydrogen-Oxygen Engines

Kerosene-oxygen fuel on which the RD-107, RD-108 and F-1 engines described above operate, is used extensively in space rockets. The velocity of the jet blast reached and exceeded 3,000 m/s with development of this fuel. Development of liquid rocket engines in the middle of the 1960's, which operate on oxygen-hydrogen fuel which exceeds approximately 30 percent the specific impulse of oxygen-kerosene fuel, was of great importance for further development of cosmonautics.

Although oxygen-hydrogen fuel was proposed as early as 1903 by Tsiolkovskiy, it did not find application for a long time for reasons related to the specific properties of hydrogen. As is known, liquid hydrogen is 14 times lighter than water and boils at 20°K.* Mixtures of hydrogen and air are extremely flammable and explosive. For example, the energy of the electrostatic discharge which we sometimes feel upon touching a door handle is tens and hundreds of times greater than the energy required to ignite a hydrogen-air mixture. In this regard production of inexpensive liquid hydrogen in large quantities was problematical, design and operation of liquid hydrogen systems were related to complex engineering problems and the fuel tanks for liquid hydrogen were too heavy.

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*Let us point out for comparison that liquid oxygen is 14 percent heavier than water and boils at 90°K.

Oxygen-hydrogen fuel is now used on the upper stages of space rockets, where it produces the greatest effect. An example of this is the universal "Centaur" stage used on space rockets of the "Atlas" and "Titan-3" families and also the second and third stages of the "Saturn-5" rocket. The fuel tanks of these rocket stages, designed to distribute liquid hydrogen, are gigantic thermos bottles whose metal walls are covered with heat-insulating polymer materials. The heat insulation used in the tanks of the "Saturn-5" rocket is shown in Figure 9 as an example. Channels through which gaseous helium is delivered during launching of the rocket to remove explosive vapors from the insulation which may be accumulated there, are provided in this insulation.

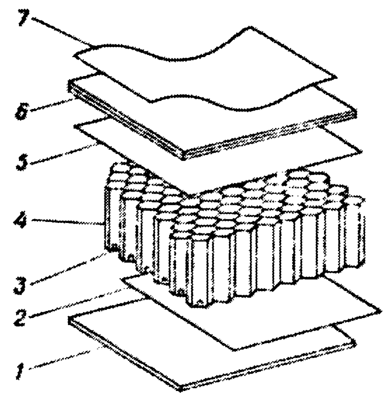

Figure 9. Wall of Liquid Hydrogen Fuel Tank (the Second Stage of the "Saturn-5" Rocket): 1 -- power sheathing (aluminum alloy); 2 and 5 -- adhesive layer; 3 -- channels for passage of helium; 4 -- honeycomb structure (polyurethane material); 6 -- nylon-phenol layer; 7 -- airtight coating (synthetic tedlar material)

The heat insulation adds to weight of the oxygen-hydrogen stages. Since the oxygen-hydrogen fuel is three times lighter than oxygen-kerosene fuel, it requires three times more volume for its distribution with the same mass. The total weight of the rocket stage design required for 1 kg of fuel is 40 percent greater for oxygen-hydrogen fuel than for oxygen-kerosene fuel. This disadvantage is compensated with an excess of high specific impulse of oxygen-hydrogen engines. With equal launch mass, a space rocket operating on oxygen-hydrogen fuel is capable of inserting three times more payload into orbit than a rocket operating on oxygen-kerosene fuel. The use of this fuel in the upper stages of the "Saturn-5" rocket permits low circular geocentric orbital injection of up to 140 t and injection onto a lunar flight trajectory of up to 48.5 t of payload.

Along with high efficiency, oxygen-hydrogen fuels have a number of other advantages, among which should be noted low combustion temperature (200°C lower than for oxygen-kerosene fuel) and nontoxicity of both the fuel itself and its combustion products (which are a mixture of water vapor and gaseous hydrogen).

Let us now dwell on the RL10 and J-2 engines used in the "Centaur" stage and on the upper stages of the "Saturn-5" rocket, respectively.

The RL10 engine, developed by the Pratt-Whitney Company, develops a thrust of 6.8 t. It is obvious from the schematic diagram of the engine shown in Figure 10 that it is a single-chamber liquid rocket engine with pump feed of fuel. However, unlike other liquid rocket engines with turbopump assemblies, there is no gas generator in the RL10: the turbine is rotated by the gas produced upon heating of liquid hydrogen in the cooling channel of the chamber. The gaseous hydrogen temperature is only -70°C; it enters the chamber after the turbine, where it burns with liquid oxygen at a pressure of approximately 28 atm (the mixture is ignited by an electric spark plug). The design temperature during the initial moment is sufficient to convert the hydrogen to a gas, which provides spinning of the turbine. The simple schematic diagram of the RL10 engine is explained by the exceptionally high thermodynamic characteristics of hydrogen

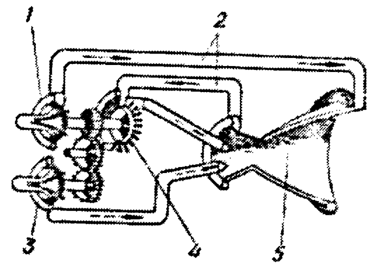

Figure 10. Schematic Diagram of RL10 Engine: 1 -- fuel pump; 2 -- fuel lines; 3 -- oxidizer pump; 4 -- turbine; 5 -- chamber

The RL10 is the best liquid rocket engine in specific impulse, which is equal to 4,360 m/s. Two of these engines are installed on gimbal suspensions on the "Centaur" stage. This stage is used extensively to launch automatic interplanetary stations. In 1972 a "Atlas" rocket with a "Centaur" stage imparted solar escape velocity to the spacecraft (the "Pioneer-10" automatic interplanetary station for investigation of Jupiter). Beginning in 1965, more than 100 RL10 engines have been used in space flights without a single failure.

The J-2 oxygen-hydrogen engine, developed by the Rocketdyne Company, develops a thrust of 104 t. It is single-chamber with pump feed of fuel and has the characteristic feature that separate turbopump assemblies, each of which consists of a pump and turbine, are provided for the oxidizer and fuel. A single gas generator to which approximately 2 percent of the fuel expended through the engine is delivered, is provided for the two assemblies. The produced gas sets the two turbines into rotation sequentially, after which it is ejected into the nozzle of the tubular chamber through slots between the tubes through which the fuel flows. The liquid rocket engine is fired and cut off (as in the RL10) by valves controlled by gaseous helium. The turbopump assemblies are turned during launching by gaseous hydrogen delivered from a special tank.

The J-2 engine develops a specific impulse of 4,170 m/s at a combustion chamber pressure of 55 atm. The liquid rocket engine, weighing 1,600 kg, is attached secured to the rocket or to the gimbal suspension. Five of these engines are installed on the second stage of the "Saturn-5" rocket and one is installed on the third stage.

Liquid Rocket Engines Operating on High-Boiling Fuel

Along with oxygen-kerosene and oxygen-hydrogen fuels containing liquefied gases and called cryogenic fuels, high-boiling fuels whose components are liquids under ordinary external conditions, are used extensively in space rockets. These fuels are inferior to the oxygen-kerosene pair in specific impulse, but exceed it in density. The basis for the use of high-boiling fuels on large scales in rocket technology was laid by development of long-range ballistic missiles operating on these fuels at the end of the 1950's and beginning of the 1960's. The problems of operating the rockets and the compatibility of fuels with the design materials, related to toxicity and chemical aggressiveness of high-boiling fuels, were resolved successfully.

One of the first liquid rocket space engines employing high-boiling fuels was the RD-214 engine, used since 1962 in the first stage of the "Kosmos" series of rockets. The RD-214 develops a thrust of 74 t and operates on a nitric acid oxidizer (AK) -- a mixture of nitrous oxides and nitric acid -- and the by-product of kerosene conversion. Modern high-boiling fuels in which the oxidizer is nitrogen tetroxide (AT) and the fuel is asymmetrical dimethyl hydrazine (NDMG) or a mixture of it with hydrazine called aerozine (AK-NDMG fuel occupies an intermediate position), are more efficient. The disadvantages of high-boiling fuels containing NDMG and aerozine are also their self-combustibility upon contact of the fuel components. The presence of these fuels permits development of liquid rocket engines simple in design and reliable in operation with a low specific weight.

Engines which operate on AK-NDMG fuel include the single-chamber liquid rocket engine with thrust of 7.3 t of the American Bell Company, used since 1959 on the universal high-altitude "Agena" rocket stage, and the RD-216 engine, developed by GDL-OKB in 1958-1960 and used on the first stage of one of the rockets of the "Kosmos" series. The RD-216 consists of two identical liquid rocket engines having a common launch system. Liquid rocket engines are two-chamber with turbopump assemblies located between the chambers in the region of their minimum cross-section. Each liquid rocket engine develops a thrust of 88 t and has a specific impulse of 2,860 m/s at chamber pressure of 74 atm.

Among the numerous liquid rocket space engines operating on nitrogen tetroxide fuels, let us indicate here only the powerful engines of the American Aerojet Company, used in the first and second stages of the "Titan-3" rocket (in the second and third stages, respectively, in the variant of the rocket with launch RDTT).

The first stage engine of the "Titan-3" space rocket consists of two single-chamber LR87-AJ-9 engines which develop a total thrust of 215 t and specific

impulse of approximately 2,810 m/s at chamber pressure of 55 atm. This engine is the best among foreign liquid rocket engines in specific weight -- 6.9 kg/t. A single-chamber LR91-AJ-9 engine with thrust of approximately 45 t and specific impulse of 3,040 m/s, operating at pressure of 58 atm, is installed in the second stage of the "Titan-3." The spent turbine gases in this engine are exhausted into a rotary nozzle, providing control of the rocket position with respect to the longitudinal axis. When the considered engines are started, the turbopump assemblies are turned by the powder gases from the pyrostarters installed on the turbine housings.

The described liquid rocket engines of the Aerojet Company were used specifically in 1965-1966 on the modified "Titan-2" intercontinental ballistic missile, which launched the 10 "Gemini" spacecraft.

The Engines of the "Kosmos" and "Proton" Space Rockets

Developing ever more efficient rocket fuels, the designers of liquid rocket engines at the same time attempted to utilize more fully the chemical energy contained in the fuels, i.e., to convert this energy with the highest possible efficiency to the kinetic energy of the engine jet.

The greatest advances in increasing the efficiency of liquid rocket engines were achieved in the USSR. Pressures several times greater than those for the RD-107 and RD-108 became possible in the soldered-welded design chambers of GDL-OKB, now used everywhere in Soviet liquid rocket engines. The new models of Soviet liquid rocket space engines were designed for ever higher chamber pressure. Liquid rocket engines operating at chamber pressure of 75-80 atm appeared 5 years after development of the first space engines. Among them were the RD-216 and RD-119 engines. Data on the RD-216 were given above and now let us devote a few lines to the RD-119 engine.

The RD-119 (Figure 11) has been used since 1962 in the second stage of one of the rockets of the "Kosmos" series. This engine operates on the oxygen-asymmetrical dimethyl hydrazine fuel pair and develops a thrust of approximately 11 t. The RD-119 is inferior in specific impulse (3,450 m/s) only to oxygen-hydrogen engines. The high specific impulse of the engine is produced by an efficient fuel, high chamber pressure (80 atm), improved mixing of the fuel prior to ignition and high degree of gas expansion in the chamber (to a pressure of approximately 0.06 atm).

The RD-119 engine is a single-chamber with pump feed of the fuel. The gas for driving the turbines is produced by thermal expansion of the fuel in the gas generator; the temperature required for the beginning of expansion is provided by combustion of a powder charge located in the gas generator (this charge is also used for initial turning of the turbopump assembly). The spent turbine gas flows through fixed steering nozzles equipped with gas distributors, providing control of the flight direction of the rocket stage and its position. The rotational speed of the turbine and the fuel pumps is 21,000 rpm. Titanium and aluminum alloys are used extensively in the design of liquid rocket engines; because of this the RD-119 is related

by specific weight (15.6 kg/t) to the best liquid rocket space engines with thrust of several tons.

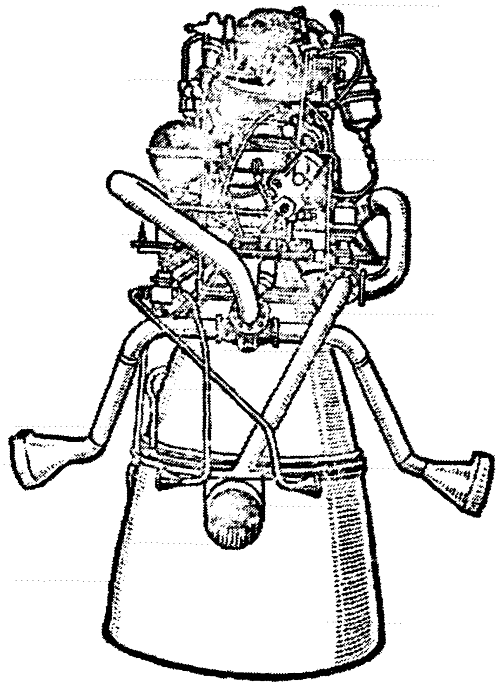

Figure 11. RD-119 Liquid Rocket Engine

The chamber pressures of the RD-119 and RD-216 are close to the maximum pressure for liquid rocket engines, in which the spent turbine gas is ejected into the exhaust pipes or into the chamber nozzle. The fact is that the design strength of the turbine which rotates the fuel pumps limits the gas temperature in the gas generator to approximately 800°C. Both the temperature and pressure in all the previously mentioned engines are reduced significantly after the gas passes through the turbine (for example, temperature is equal to 600°C at pressure of 1.25 atm in the RD-216 engine).

As we already know, gas with such low parameters may not produce a high specific impulse. Thus, although the portion of the liquid rocket engine fuel which is expended to drive the turbopump assembly does develop thrust, the specific impulse of the engine is below that which would be produced if all the fuel were ignited in the combustion chamber. For example, the specific impulse for the RD-216 engine is 1-1.5 percent below the specific impulse of the chamber.

The head of the fuel pumps and consequently their output must be increased to increase the chamber pressure. To do this, the flow rate of the gas rotating the turbine must in turn be increased. Thus, the fraction of engine fuel expended through the gas generator increases as the chamber pressure increases, which leads to slowing of the increase of specific impulse for the liquid rocket engine of the described scheme and also to a decrease of this parameter.

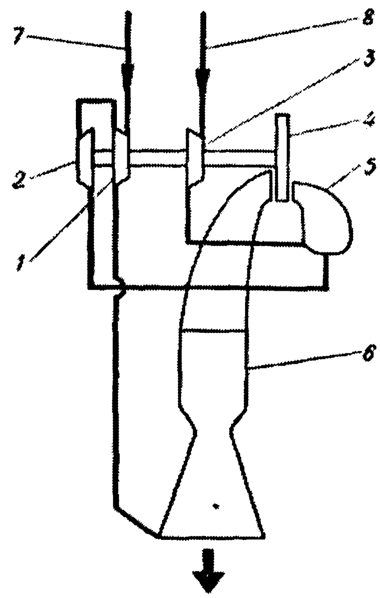

Figure 12. Diagram of Liquid Rocket Engine With Afterburning: 1 -- first stage of fuel pump; 2 -- second stage of fuel pump; 3 -- oxidizer pump; 4 -- turbine; 5 -- gas generator; 6 -- chamber; 7 -- fuel; 8 -- oxidizer

A significant increase of chamber pressure in liquid rocket engines developed after the RD-216 and RD-119 engines was achieved by developing an engine scheme with afterburning (Figure 12). The greater portion of all the consumed fuel enters the gas generator in these engines. The gas pressure in the gas generator is approximately double that of the gas in the combustion chamber. The spent turbine gas enters the chamber mixing head for afterburning with the remaining part of the fuel at a pressure of 150 atm or more.

The indicated engines, characterized by high specific impulse and efficiency, have been used extensively since the middle of the 1960's in Soviet rocket technology. Specifically, they have been used since 1965 on all stages of the "Proton" space rocket. Several RD-253 engines designed by GDL-OKB, which significantly exceed the thrust of the RD-107 and RD-108 engines, were installed on the first stage of this rocket. Some data on the RD-253 engine are presented below.

The RD-253 is a single-chamber engine operating on high-boiling calorific self-igniting fuel. The major portion of the fuel after the pumps (approximately 75 percent) enters the gas generator welded to the turbine housing and the remaining part is delivered to the regenerative cooling channel of the chamber. The gas after the turbine is delivered through a gas line to the combustion chamber, where it is burned with the liquid component that has passed through the chamber cooling channel.

The combustion chamber pressure of the RD-253 is approximately three times greater than the corresponding pressure for the RD-108. For reliable cooling of the chamber, its fire wall is protected by a refractory ceramic coating

and gas-liquid film formed by delivery of the fuel component from the cooling channel to the wall through openings in it. The total power of the turbopump assemblies of all the RD-253 engines contained in the propulsion plant exceeds 150,000 hp; a power of 100 hp is concentrated in a single kilogram of weight of the turbopump assembly (this assembly operates at a speed of 14,000 rpm). The maximum fuel pressure in the main lines of the engine reaches 400 atm. The design reliability of the engine at such high pressure is provided by extensive use of welding to join the assemblies and subassemblies.

The liquid rocket engine is fired by "gravity flow" of the fuel (similar to the F-1 engine) without the use of special starting devices. The use of self-igniting fuel eliminates the need for an ignition system. The liquid rocket engine is fired and cut off by pyrovalves of simple design. A regulator and choke operating from electric drives are installed in the main lines to control the thrust and ratio of fuel component flow rates during flight of the engine. Supercharging assemblies, which are small gas generators in which gases are produced for supercharging the rocket fuel tanks, are provided in the engine. The assemblies for attaching the engine to the rocket provide rotation of it in a plane parallel to the longitudinal axis to control the flight direction of the rocket and its position. The engine is similar to the RD-107 in height but has considerably smaller overall dimensions in horizontal measurement. The specific weight of the RD-253 engine is 7.7 kg/t.

Development of liquid rocket engines for the "Proton" space rocket was an important achievement of space rocket technology. Such outstanding events as near-earth orbits of the automatic scientific stations "Proton" with mass up to 17 t and the launch of automatic recoverable stations "Zond" for orbiting the moon, delivery of lunokhods to the moon, launch of the automatic interplanetary stations which took samples of lunar soil, which landed on Mars and which became satellites of these celestial bodies, are related to the use of this rocket.

Engines With Pressure Feed of Fuel

Besides liquid rocket engines with pump feed of fuel, different models of which were described above, a few models of liquid rocket engines with pressure feed are used in space rockets. These engines, which operate on high-boiling self-igniting fuels, are used primarily in the upper rocket stages whose propulsion plants develop a thrust of several tons and may be fired repeatedly during flight.

Of greatest interest among the indicated engines is the AJ10-138 engine designed by the Aerojet Company. Two of these engines with a total thrust of 7.2 t were installed on the last stage of one of the space rockets of the "Titan-3" family and were designed for repeated firing for 6.5 hours of orbital flight. The indicated rocket stage, called "Transstage," is used to solve numerous problems, including simultaneous insertion of several payloads into different orbits (up to eight satellites were injected).

Reliable operation of the AJ10-138 engine and of the propulsion plant as a whole was provided to a large extent by the type of fuel used and the selected schematic diagram of the engine. Nitrogen tetroxide and aerozine, which as we already know provide design simplicity of liquid rocket engines and which are easily stored without losses under space flight conditions, are used as the fuel. The fuel is delivered to the chamber at helium pressure. The chamber operates at a pressure of 7 atm and is made without a regenerative cooling channel. For this reason the pressure in the fuel tanks hardly exceeds the chamber pressure and the weight of the fuel tanks is sufficiently small.

The chamber of the described liquid rocket engine consists of a removable mixing head which contains several hundred openings for atomization of the fuel and is made of aluminum alloy, and a housing designed for cooling by ablation and radiation of heat into the surrounding atmosphere. Mass is removed from the inner surface of the chamber by the hot gas flow during ablation. The combustion chamber and the initial section of the nozzle are cooled in this manner (the remaining part of the nozzle is cooled by radiation). This part of the chamber is manufactured on a production mandrel. An ablating polymer material -- refrasil fibers impregnated with a rubberized phenol resin, with subsequent vulcanization of these fibers, is initially applied to the mandrel. A layer of heat insulation -- asbestos felt impregnated with phenol resin, which is also vulcanized, is then applied and finally an outside power sheathing -- fiberglas impregnated with epoxide resin, is applied to this layer after which the mandrel is removed. The power sheathing is designed for a temperature of not more than 180°C.

The total operating time (operating life) of liquid rocket engines is limited by the permissible ablation of the material and comprises 500 s. This operating time is provided by creation of a protective gas-liquid film near the chamber fire wall by atomization of the fuel through peripheral openings of the mixing head. The output section of the nozzle, cooled by radiation, is a thin-walled shell manufactured from refractory alloys: niobium and titanium. This shell is heated to 900-1100°C during operation of the engine.

The engine is fired by fuel control and auxiliary solenoid valves. The engine is installed in a gimbal suspension on the rocket stage and is deflected during flight by electromechanical drives.

The comparatively high specific impulse (3,000 m/s) of the AJ10-138 engine at low gas pressure in the combustion chamber was achieved by expanding the gases in the jet nozzle to a very low pressure. Based on data on the working principle of liquid rocket engines contained at the beginning of the pamphlet, one may conclude that the considered engine has large overall dimensions and weight. Actually, its height is 2.0 m, diameter is 1.2 m and specific weight is approximately 25 kg/t. Thus, the AJ10-138 engine, which develops a thrust of less than 4 t, is comparable in overall dimensions to the LR91-AJ-9 engine, whose thrust is more than 45 t; the AJ10-138 is considerably inferior to the RD-119 engine in weight ratio.

The Main Liquid Rocket Engines of Spacecraft

The conditions under which these vehicles are found -- weightlessness, deep vacuum, space cold and the thermal radiation of the sun and planets -- have a significant effect on operation of space propulsion plants with liquid rocket engines and largely determine their design and characteristics.

For example under weightlessness the fuel in the tanks is randomly mixed (if appropriate measures are not taken) with gas bubbles used to pressurize the tanks. If gas enters the engine when it is being fired, the engine operating process is disrupted with probable failure of its main assemblies. To prevent this, special components which separate the gas and liquid are provided in the fuel tanks of spacecraft engines: elastic bags, elastic membranes, bellows and so on. The gas and liquid fuel may also be separated by special design of the intake fuel devices (reticular or capillary) and by creation of artificial gravity by auxiliary engines* when the engines are fired.

The deep vacuum under which liquid rocket engines of spacecraft operate is the reason that the surfaces of the engine components leave gas molecules adsorbed on the surfaces and also particles of structural materials, coatings and lubricants. Variation of the frictional properties of the surfaces caused by these processes may cause spontaneous welding of the contacting metal parts. Subsequent settling of the removed metal particles on the components of the engine electrical equipment may lead to short circuiting.

All these undesirable phenomena place specific limitations on selection of engine materials, engine configuration and its arrangement on the spacecraft.

Moreover, the negative effect of cosmic radiation on the materials and working products used should be taken into account in some cases when designing space propulsion plants with liquid rocket engines.

The outlined concepts are related to all liquid rocket engines operating in space. The characteristics of specific models of these engines are determined by the problems for solution of which they are designed.

Characteristics of the Main Liquid Rocket Engines of Spacecraft

Liquid rocket engines which provide significant variation of spacecraft velocities, usually related to transferring the vehicles from an artificial satellite orbit of the planet to a flight trajectory toward another planet or into space, transfer of vehicles from flight trajectory to artificial satellite orbit, landing on a planet and takeoff from it, transfer from an intermediate to a different orbit and so on are discussed in this chapter.**

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*The problem of separating the fuel and gas must also be solved for liquid rocket engines of the upper stages both during the first and during subsequent firings of the engines.

**Operations which require multiple firing of propulsion plants are frequently carried out by RDTT.

Significant variation of spacecraft velocity occurs in a number of cases during maneuvering operations and correction of flight trajectory.

According to the resolved problems, the main spacecraft engines may be booster, retrofire, correcting, takeoff, landing and so on. One engine may also fulfill several functions, including auxiliary functions.

Individual models of the considered liquid rocket engines differ considerably between each other in design and characteristics: they are made both with pump and with pressure feed of fuel. The thrust of these engines encompasses a range from tens of kilograms to several tons and may be both regulated and unregulated; they may operate continuously for tenths of a second and several thousand seconds; the number of firings of these engines varies from one to several tens. Faultless operation of all these engines in space is provided to a large degree by the use of high-boiling self-igniting bipropellant or of high-boiling monopropellant in them.

Liquid rocket engines with pump feed of bipropellant having regeneratively cooled chambers are used extensively in Soviet spacecraft. Engines only with pressure delivery of fuel, whose chambers are frequently cooled by ablation or radiation and also by creation of a protective gas-liquid film, are used in foreign spacecraft.

Some models of Soviet and American engines are presented below which give an idea about the main spacecraft engines. The Soviet engines were developed under the leadership of A. M. Isayev.

The Engine of the First Spacecraft

This engine was a single-chamber liquid rocket engine with pump feed of high-boiling self-igniting fuel, consisting of a nitrous oxide oxidizer and amine-based fuel. The engine developed a thrust of 1,614 kg and specific impulse of 2,610 m/s at a combustion chamber pressure of approximately 57 atm. The engine was part of the TDU-1 retrofire rocket, which developed a thrust to transfer the spacecraft from artificial earth satellite orbit to a reentry trajectory. The engine was located in the center of the toroidal fuel tank pod. This configuration of the propulsion plant provided compactness and low weight of the design.

Fuel feed to the engine was provided during firing in orbital flight by elastic membranes installed in the tanks and which separate the fuel and pressurization gas (nitrogen). The conditions for normal ignition of the fuel in the combustion chamber were created by insulating the combustion chamber cavity from the surrounding medium by a thin muffler welded into the jet nozzle in the region of minimum cross-section. This muffler ejected the gases formed in the combustion chamber from the beginning of engine operation.

The TDU-1 was used on the first Soviet spacecraft "Vostok," piloted by Yu. A. Gagarin and G. S. Titov, and in somewhat modified form on the subsequent spacecraft of this type and also on the "Voskhod" spacecraft.

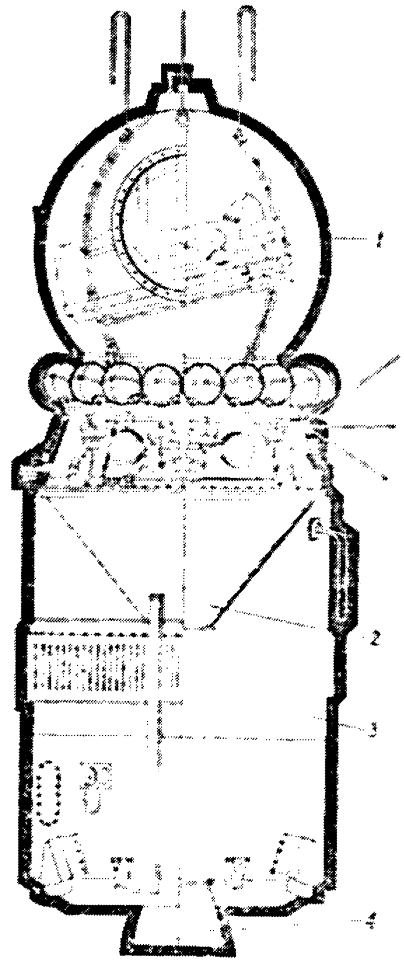

Figure 13. The "Vostok" Spacecraft With Last Rocket Stage: 1 -- recovery capsule; 2 -- retrofire rocket; 3 -- last stage of "Vostok" rocket; 4 -- liquid rocket stage

The TDU-1 joined to the "Vostok" spacecraft is shown in Figure 13.

The Engines of the Automatic Interplanetary Stations "Luna," "Venera" and "Mars"

The successful flights of Soviet automatic interplanetary stations to investigate the planets of the solar system were provided by propulsion plants with liquid rocket engines.

Launches of automatic interplanetary stations to the moon were begun in 1959. The problem of reaching the moon was solved initially. In this regard the flight of the first three AMS of the "Luna" type occurred on a ballistic trajectory and they were not equipped with their own engines. Solution of more complex problems in investigation of the moon, which science faced, required development of special propulsion plants for automatic interplanetary stations and launch of vehicles from earth parking orbits by firing the engine of the last stage of the space rocket.

The AMS which made the world's first (1966) soft landing on the moon and which went into circumlunar orbit were equipped with the KDTU-1 correcting-retrofire propulsion plant. It included a single-chamber liquid rocket engine with pump feed of the same fuel as used in the engine of the "Vostok" spacecraft. The liquid rocket engine developed a thrust of 4,640 kg and specific impulse of 2,720 m/s at a combustion chamber pressure of approximately 64 atm.

Fuel delivery to the liquid rocket engine without gas inclusions during starting was provided by reticular separators in the tanks which utilize the property of the surface tension of liquid in the cells of a find metal mesh. The engine was fired twice during flight. The first firing provided correction of the flight trajectory of the AMS toward the moon and the second provided braking to transfer the AMS into an artificial lunar satellite or to reduce the approach velocity of the AMS to the moon to a safe value.

The given flight direction and the specific position of the AMS in space were provided by steering nozzles into which was fed the spent turbine gas. These same nozzles provided (with a nonoperating chamber) the final approach velocity of the AMS to the lunar surface.

Later versions of the "Luna" type AMS delivered lunar soil specimens to the earth and automatic self-propelled vehicles "Lunokhod-1" and "Lunokhod-2" to the lunar surface. Successful accomplishment of these complex problems was provided by development of a new standardized attitude braking rocket and, moreover, a liquid rocket engine for the landing stage of the AMS ("Luna-16" and "Luna-20"). The propulsion plant operated on high-boiling self-igniting fuels containing asymmetrical dimethyl hydrazine and consisted of two autonomous pods (the main and a low-thrust pod). The main pod consists of a single-chamber liquid rocket engine with pump feed of fuel and was designed to operate in three different thrust modes (from 1,930 to 750 kg) and multiple firing (up to 11 times). The low-thrust pod contains a two-chamber liquid rocket engine for one-time firing with pressure delivery of fuel (by helium); it is also designed to operate in three different thrust modes from 210 to 350 kg). The specific impulse for the liquid rocket engine of the main pod is 3,080 m/s and the total operating time is 650 s; these parameters are equal to 2,490 m/s and 30 s, respectively, for the liquid rocket engines of the low-thrust pod.

The propulsion plant is all-welded and the liquid rocket engine chamber of the main pod is the load-bearing member to which all the other components are attached.

The engine of the takeoff stage of the AMS "Luna" is a liquid rocket engine with pump feed of the same fuel on which the low-thrust engine pod operates. The engine develops a thrust of 1,920 kg at combustion chamber pressure of approximately 94 atm. The launch of the takeoff stage of the AMS "Luna-16" is shown on the back cover.

The leading role in the study of Venus belongs to Soviet science. Launches of AMS to this planet were begun in 1961 in the USSR. The flight schemes initially provided entry of AMS into the atmosphere of Venus at escape velocity, with separation of the descent capsules, braked by aerodynamic drag, from the AMS. The flight trajectory of the AMS to Venus was provided by a vernier rocket with liquid rocket engine. This liquid rocket engine (Figure 14) was one with pressure feed of fuel operating on a nitric acid oxidizer and asymmetrical dimethyl hydrazine. The engine was installed in a gimbal suspension; it developed a thrust of 200 kg and specific impulse of 2,670 m/s at chamber pressure of 12 atm.

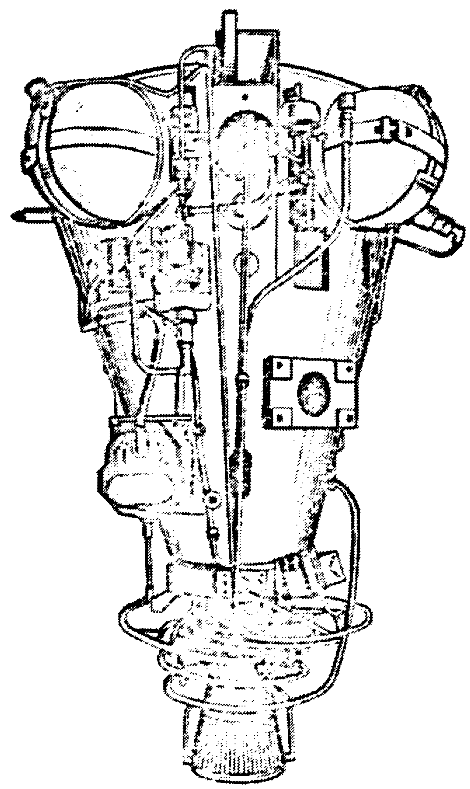

Figure 14. Vernier Engine of First AMS "Venera"

The described engine provided flight of "Venera" stations whose descent capsules first made a smooth descent in the atmosphere of Venus (1967) and which transmitted information from the surface of this planet (1970). Similar liquid rocket engines were used in the space research program using AMS of the "Zond" series, the last variants of which circumnavigated the moon with subsequent return to earth. These same engines were used on the "Molniya-1" communications satellites and the first AMS launched toward Mars.

The latest achievements of Soviet science and technology in investigation of the planets of the solar system are related to development of more improved propulsion plants with liquid rocket engines. Thus, vernier-braking

rockets with pump feed of fuel, capable of reliable operation during the many months of spaceflight in different thrust modes (from approximately 1,000 to 1,900 kg) and during multiple firing (up to seven times), were developed for the AMS "Mars." The combustion chamber pressure of these liquid rocket engines exceeds 95 atm and the specific pulse reaches 3,090 m/s.

The "Soyuz" Spacecraft Engines

The "Soyuz" spacecraft is equipped with an approach-correcting propulsion plant which includes two liquid rocket engines -- a main and backup. The main engine creates thrust to correct the orbit of the "Soyuz" spacecraft, its maneuvers during approach to another spacecraft and to brake the spacecraft during descent from artificial earth satellite orbit. The backup engine operates in case of failure of the main engine and during impermissible deviations in operation of the auxiliary engines (with which the spacecraft is also equipped).

Both engines of the propulsion plant have pump feed .r fuel (a nitric acid oxidizer and asymmetrical dimethyl hydrazine) with unregulated thrust of approximately 415 kg. The specific pulse of the main engine is almost 2,770 m/s, the chamber pressure is 40 atm and the nozzle output pressure is 0.04 atm. Unlike the two-chamber backup engine, this engine is a single-chamber type. The main engine is equipped with steering nozzles to which the spent turbine gas is fed.

This engine is the first liquid rocket engine with pump feed of fuel which permits reliable multiple starts and operation both over a long period of time (several hundred seconds) and in the short pulse mode (lasting tenths of a second).

The "Apollo" Spacecraft Engines

The "Apollo" spacecraft on which flights of American astronauts to the moon were made during the period 1969-1972, is equipped with three main liquid rocket engines according to the spacecraft diagram shown in Figure 15.

An AJ10-137 liquid rocket engine of the Aerojet Company is installed in the service module of the spacecraft. It provides correction of the flight trajectory to the moon, insertion of the spacecraft into circumlunar orbit, transfer of the crew module from this orbit to a flight trajectory toward earth (with return of the astronauts) and trajectory correction. The liquid rocket engine develops a thrust of 9.3 t, its weight is 370 kg and height is 3.9 m.

An LMDE liquid rocket engine of the Thompson-Ramo-Wooldridge Company is installed on yhr landing stage of the lunar module. The thrust of its engine is regulated in the range of 4.5-0.45 t, its weight is 170 kg and height is 2.5 m. An RS18 liquid rocket engine of the Rocketdyne Company is installed on the takeoff stage of the module. The thrust of this engine is 1.6 t, weight is 90 kg and height is 1.3 m. The engines of the lunar module are shown in Figure 16.

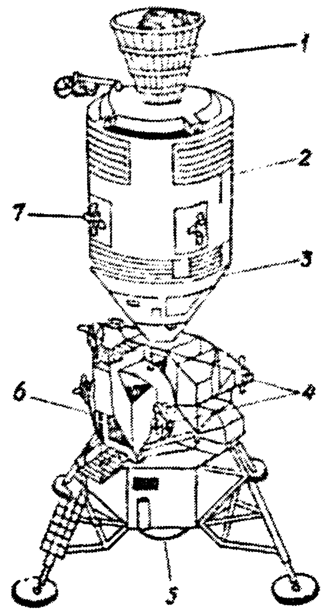

Figure 15. "Apollo" Spacecraft: 1 and 5 -- main liquid rocket engines; 2 -- service module; 3 -- return vehicle (crew module); 4 and 7 -- auxiliary liquid rocket engines of jet control systems; 6 -- lunar module

All three liquid rocket engines are based on common principles, the same as the AJ10-138 engine described above in the section devoted to liquid rocket space engines: they are single-chamber engines with pressure feed (by helium) by fuel (nitrogen tetroxide-aerozine). The engines operate at low chamber pressure (7-8.4 atm) and the chamber is designed for ablation and radiation cooling and the use of a gas-liquid fuel film. The comparatively high specific impulse of the liquid rocket engines (from 2,980 m/s for the LMDE to 3,080 m for the AJ10-137) is achieved by large jet nozzles. The long operating life (from 10 minutes for the RS18 to 15 minutes for the LMDE*) is provided by reducing the gas temperature in the combustion chamber by selecting the corresponding ratio of fuel components.

The AJ10-137 and LMDE liquid rocket engines are installed in gimbal suspensions and may be deflected by electric drives. All three engines are fired by oxidizer and propellant fuel values combined into single pods. These valves and the other components of the liquid rocket engine control system are reserved to increase the engine reliability. If the propulsion plant with AJ10-137 engine fails, the crew may be returned to earth by the LMDE landing engine. This capability was realized successfully during the emergency flight of an "Apollo" spacecraft to the moon in 1970 ("Apollo-13").

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*The liquid rocket engines operated less than the indicated time during flights to the moon.

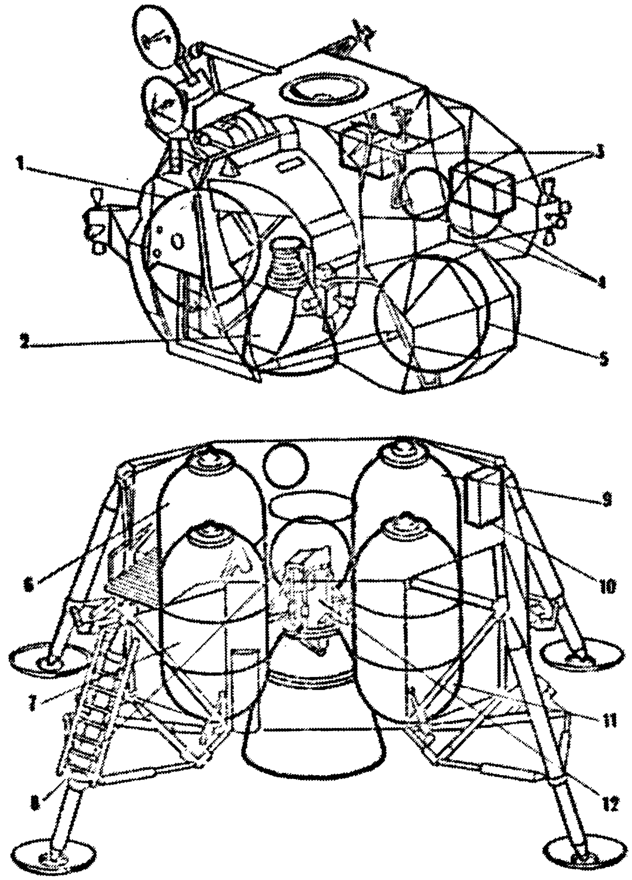

Figure 16. Arrangement of Main Liquid Rocket Engines of the Lunar Module of the "Apollo" Spacecraft (the Takeoff Stage of the Module Is Shown on Top): 1, 7 and 9 -- oxidizer tanks; 2 -- liquid rocket takeoff engine; 3 and 10 -- assemblies of the fuel tank pressurization system; 4 and 8 -- helium tanks for pressurizing the fuel tanks; 5, 6 and 11 -- fuel tanks; 12 -- liquid rocket landing engine

The AMS "Viking" Engines

The American program for investigation of Mars including landing of two AMS "Viking" consisting of orbital and descent capsules,* on the surface in July-September 1976.

The orbital vehicle is equipped with an RS2101C liquid rocket engine of the Rocketdyne Company, designed to correct the interplanetary flight trajectory of the AMS, to insert it into the orbit of an artificial Martian satellite and to correct this orbit. The indicated engine is a modification of the

*Both "Vikings" have already landed on the surface of Mars and are transmitting scientific information.

engine used for the same purposes in the AMS "Mariner-9," which photographed almost 80 percent of the Martian surface from the orbit of this planet in 1971.

The RS2101C is a single-chamber liquid rocket engine with pressure feed of a high-boiling self-igniting fuel (nitrogen tetroxide-monomethyl hydrazine). The engine is designed for 24 firings and a total operating time of almost 1 hour (the length of one firing is 0.4 seconds to 45 min). The engine develops a thrust of approximately 140 kg and specific impulse of more than 2,850 m/s at chamber pressure of 8 atm. The engine is installed in a gimbal suspension. The weight of the engine is 8.2 kg, height is 0.6 m and diameter is 0.3 m.

A soft landing of the descent capsule of the "Viking" spacecraft is provided by three MR-80A liquid rocket engines of the Rocket Research Company, which operate on a monopropellant — hydrazine. The indicated engine is shown in Figure 17. Its thrust is regulated over a range from 270 to 40 kg with operating time up to 10 min. A large number of jet nozzles (18) is provided in the engine to reduce the degree of damage and contamination of Martian soil by the gas jet of the working engine.

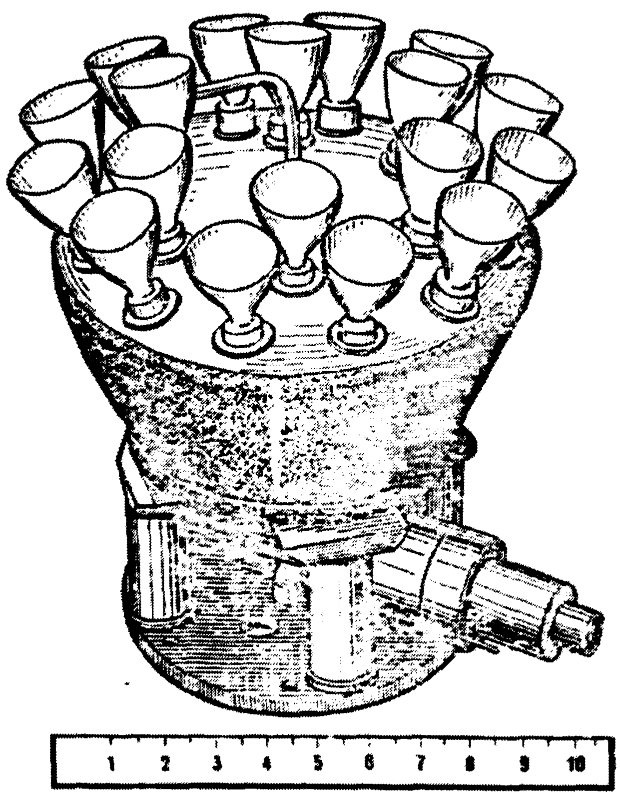

Figure 17. Liquid Rocket Engine of Descent Vehicle of the "Viking" Spacecraft (One Scale Division Corresponds to 1.5 cm)

The indicated fact is related to one of the main problems of the "Viking" program -- investigation of possible life on Mars, which presumes sampling and analysis of Martian soil specimens. This problem predetermined selection of hydrazine monopropellant (rather than bipropellants) for the liquid rocket engines of the landing module of the spacecraft. The fact is that the jet stream of the hydrazine engine has comparatively low temperature, contains no undesirable by-products and, finally, the landing module with the hydrazine-charged propulsion plant may be easily sterilized prior to launch by holding it at an increased temperature -- approximately 130°C (the vehicle is sterilized to eliminate possible transport of earth microorganisms to Mars).

Liquid rocket engines operating on hydrazine are discussed in detail in the next chapter.

Auxiliary Liquid Rocket Engines

These engines are used for orientation, stabilization of position, spacecraft and rocket stage trajectory correction, to separate fuel and pressurization gas (settling of fuel) in the fuel tanks prior to firing of the main engines and so on.

The auxiliary engines, similar to the main liquid rocket engines of spacecraft, operate on high-boiling self-igniting fuel or on monopropellant, but only with pressure feed of it. The considered engines are significantly inferior on the whole to other liquid rocket space engines in thrust level. For example, the nominal thrust of the liquid rocket engines used to control the service module of the "Apollo" spacecraft is approximately 45 kg, whereas the thrust of the main engine exceeds 9 t. Auxiliary engines with a thrust of several kilograms or less are related to so-called microrocket engines.

The most specific among all the auxiliary engines are those used for orientation and attitude control of spacecraft (the rocket stages). Several of these liquid rocket engines, oriented by different control channels (pitch, yaw and rolling) and grouped in pods, usually form the jet control system of the spacecraft together with the common fuel tanks. The liquid rocket engines of jet control systems may operate both in the continuous thrust mode (with fixed or regulated value) and in the pulsed mode in which the engine is fired periodically for a specific time. The pulse repetition rate and pulse elngth may be quite different.

A characteristic feature of liquid rocket engines used in jet control systems is the possibility of a very large number of firings, which may reach several hundred thousand.

All the indicated characteristics of auxiliary liquid rocket space engines are easily in the example of specific models of these engines which will now be considered. These engines are divided for convenience into two groups according to the type of fuel used in them. In conclusion we shall also

briefly discuss liquid rocket engines which cosmonauts may use for propulsion in open space.

Monopropellant Liquid Rocket Engines

Liquid rocket space engines of this type operate on low-water hydrogen peroxide (concentration of approximately 90 percent) or hydrazine, decomposed in the chamber of the engine in the presence of a catalyst; the formed high-temperature gas creates thrust when it flows from the nozzle.

A typical diagram of a propulsion plant with monopropellant liquid rocket engine is shown in Figure 18. The fuel is stored in an elastic bag located in the tank and enters the chamber under gaseous nitrogen or helium pressure. The fuel decomposition catalyst is located in the chamber itself. Fuel feed to the chamber is controlled by a high-speed electric solenoid valve.

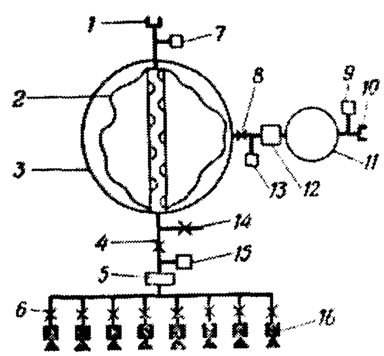

Figure 18. Schematic Diagram of Jet Control System With Monopropellant Liquid Rocket Engine: 1 and 10 -- filling valves; 2 -- elastic bag; 3 -- fuel tank; 4, 6 and 8 -- control valves; 5 -- filter; 7, 9, 13 and 15 -- drain-safety valves; 11 -- high-pressure gas tank; 12 -- pressure reducer; 14 -- drain valve; 16 -- liquid rocket engine chamber

Auxiliary liquid rocket engines operating on hydrogen peroxide were used first. Development of these engines was relatively simple, since hydrogen peroxide had already been used prior to this as a monofuel in rocket propulsion plants and as a source for producing gas to drive the turbopump assemblies. Moreover, liquid rocket engines were tested, for example, in the United States for the jet control systems of experimental rocket aircraft.

The weight and operating characteristics of monopropellant engines are determined to a great extent by the characteristics of the fuel decomposition catalyst. Potassium and silver permanganate were used as catalysts in auxiliary liquid rocket space engines. In the first case the catalyst packet arranged in the engine chamber is a mixture of carrier substance granules impregnated with potassium permanganate (Figure 19). In the second

case the catalyst packet is made in the form of a silver-coated wire frame usually manufactured from stainless steel. Liquid rocket engines with this type of catalyst packet are more improved.

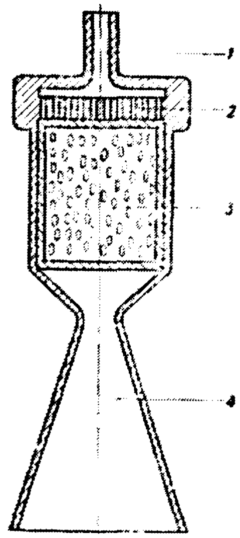

Figure 19. Liquid Rocket Engine Chamber Operating on Hydrogen Peroxide: 1 -- monopropellant feed line; 2 -- atomizer head; 3 -- catalyst packet; 4 -- jet nozzle

The gas temperature in the decomposition chamber is approximately 800°C and the specific impulse is approximately 1,400-1,600 m/s for liquid rocket engines operating on 90 percent hydrogen peroxide. Higher values of the indicated values correspond to a fuel with greater hydrogen peroxide concentration. It should be noted in this regard that the requirements on selection of the structural materials and the cleanliness of the working components coming into contact with the fuel increase as the hydrogen peroxide concentration increases to avoid spontaneous decomposition of it. The relatively low temperature of the gases formed upon decomposition of hydrogen peroxide permits manufacture of the chamber for liquid rocket engines operating on this fuel from ordinary stainless steel (by designing them for radiant cooling).

These engines (designed by the Bell Company) were used, for example, in 1962-1963 on the "Mercury" spacecraft and have been used since 1963 on the "Centaur" rocket stage. A total of 18 liquid rocket engines joined into two independent jet control systems -- automatic and manual, were used on the "Mercury" spacecraft. Three types of liquid rocket engines -- with thrust of 0.45, 2.7 and 11 kg, operating in the pulsed mode, were used in the automatic system. Liquid rocket engines of the latter two types, but with regulated thrust, were used in the manual system, which was the backup system.

Twelve auxiliary liquid rocket engines with thrust of 2.7 kg each, joined into four pods installed on the periphery of the stage, are used in the

rocket stage of the "Centaur." They provide orientation of the stage, settling of fuel in the tank and braking of the stage upon separation of the spacecraft.

Hydrogen peroxide was used extensively in the auxiliary liquid rocket space engines during the middle 1960's, after which it gradually began to be replaced by hydrazine and liquid bipropellants, which permit liquid rocket engines to produce a higher specific impulse.

The specific impulse of liquid rocket engines was increased by approximately 40 percent upon conversion from hydrogen peroxide to hydrazine. Moreover, unlike highly concentrated hydrogen peroxide, hydrazine is not subject to spontaneous decomposition. The indicated advantages of hydrazine were evaluated in 1958-1959, when development of liquid rocket engines for automatic interplanetary stations was begun in the United States.

However, the nature of hydrazine decomposition was little studied, which caused difficulties in development of a catalyst which could actively decompose the hydrazine at room temperature and tolerate multiple firings of the engine. This catalyst appeared in the United States in 1964. It was developed by the Schell Development Company and was called "Schell-405."

The active substance of the catalyst which provides the composition is iridium. It is applied to porous granules of the carrier substance, which is aluminum oxide.

Numerous models of liquid rocket space engines operating on hydrazine were developed and found application with the appearance of the indicated catalyst abroad. The range of thrusts developed by these engines is from approximately 20 g to 300 kg.

The gas temperature in the decomposition chamber of hydrazine engines is comparatively low (approximately 1000°C), which permits manufacture of the engine chamber from refractory alloys, taking into account their radiant cooling (as in the case of liquid rocket engines operating on hydrogen peroxide). A typical hydrazine-fueled engine, developed by the Rocket Research Company, one of the leading American companies in the field of developing these engines, is shown in Figure 20. Hydrazine engines for satellite control systems in which these engines are used extensively are discussed below.

Hydrazine engines are used specifically in communications satellites operating in geosynchronous orbits. These satellites are usually stabilized by rotation and several pairs of liquid rocket engines with a thrust of approximately 2 kg, which provide given angular satellite rotational velocity (usually 60-100 rpm), turning of the rotational axis and also maintaining and maneuvering of the satellite in the orbital plane, are used in them.

Liquid rocket engines may operate in both pulsed and in steady thrust modes. A typical pulsed mode includes transmission of a series of thrust pulses

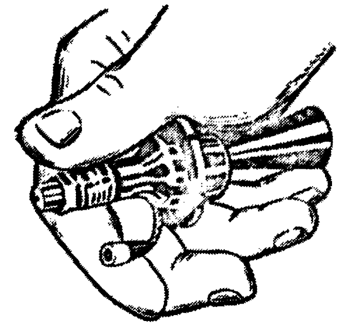

Figure 20. Hydrazine-Propelled Engine With Thrust of 0.9 kg (United States)

lasting 0.1 second with a pause of 0.9 second between them. The series may consist of several pulses and also of several hundred pulses. Their total number comprises several tens of thousands. The total length of liquid rocket engine operation in the steady thrust mode reaches several hours. Liquid rocket engines are designed to operate for several years.

For example, REA16-6 liquid rocket engines of the Hamilton Standard Company with a thrust of 2.3 kg are used in the "Skynet-2" satellite; their weight is 320 g, height is 17 cm and diameter is 3 cm (the fuel reserve for operation of these engines is only 23 kg). MRE-4A liquid rocket engines of the Thompson-Ramo-Wooldridge Company with thrust of 1.6 kg were used on the "Intelsat-3" satellite; their weight is 245 g, height is 11 cm and diameter is 2.5 cm. The fuel tanks on this satellite are installed so that the centrifugal force developed during rotation provides reliable separation of the fuel and pressurization gas, which is located in the fuel tanks themselves.

Hydrazine engines with the lowest thrust are used in satellite orientation systems stabilized in three axes. The thrust of these engines comprises less than 50 g; they are designed for an operating life up to 450,000 working pulses and may be operated up to 7 years. Four of these engines together with 16 hydrazine engines having a thrust of approximately 500 g are used, for example, in the "Fleetsatcom" communications satellite; the hydrazine is stored in fuel tanks together with the pressurization gas (hydrogen) and is separated from the latter by an elastomer diaphragm. Electric heating of the entire satellite propulsion plant is provided to maintain the catalyst packet of the liquid rocket engine at approximately 320°C. The need for heating is related to the fact that a large number of starts of hydrazine engines with a cold catalyst leads to breakdown and loss of quality. It should be noted in completing the survey of hydrazine engines that, despite the intensive development of these engines, the process of their development is primarily empirical in nature. Each type of hydrazine engine design works well only in a strictly determined mode and it is impossible to predict the extent to which it will satisfy new requirements.

Bipropellant Liquid Rocket Engines

The main advantages of propulsion plants with monopropellant liquid rocket engines is their comparative simplicity and consequently the high design reliability. These propulsion plants also have weight advantages over other jet systems within a specific range of values of total thrust impulse (i.e., multiplication of thrust by the total operating time). Bipropellant liquid rocket engines are advantageous as the required total thrust impulse increases with regard to their higher specific impulse.* Moreover, no catalyst, which determines to a great extent the dynamic characteristics of the working process and which limits the operating life of the engine, is required to operate these liquid rocket engines.

Auxiliary bipropellant liquid rocket engines have now become as common in cosmonautics as monopropellant engines. Their development was accelerated to a large extent by manned space flight programs. Specifically, they played an important role in the "Apollo" program. A total of 52 auxiliary liquid rocket space engines of four types with thrust from 34 to 68 kg was used in the "Saturn-5-Apollo" space rocket system. They operate at all stages of space flight, beginning with launch of the "Apollo" spacecraft to the moon and ending with return of the astronauts to earth.

The R-4D liquid rocket engines of the Markward Company, installed on the body of the spacecraft service module, are shown in Figure 21. This module has four self-contained controlled subsystems with their own fuel tanks and a compressed gas tank for fuel feed. The auxiliary liquid rocket engines are grouped in four pods. Each pod weighs 18 kg and consists of four R-4D engines arranged in a cross shape.

The indicated engine pods are part of the corresponding control subsystems which are almost identical and form in aggregate a system which provides orientation and stabilization of the service module and spacecraft as a whole, maneuvers of the module and other operations. Operation of the subsystems is matched, which is required for uniform delivery of fuel. Reliable spacecraft control is provided even if individual components of the system fail, including failure of the subsystems.

The R-4D engines were also used in the lunar module control system of the "Apollo" spacecraft. This system also contains four liquid rocket engine pods, but the engines are distributed between two self-contained subsystems of eight engines each. The auxiliary engines may be switched to feeding from the fuel tanks of the main propulsion plant if required.

The lunar module jet control system is installed on its takeoff stage (see Figures 15 and 16) and provides separation of the lunar module from the service module, orientation, stabilization and maneuvering and also other

*Compressed-gas jet systems may be used at small values of total thrust impulse.

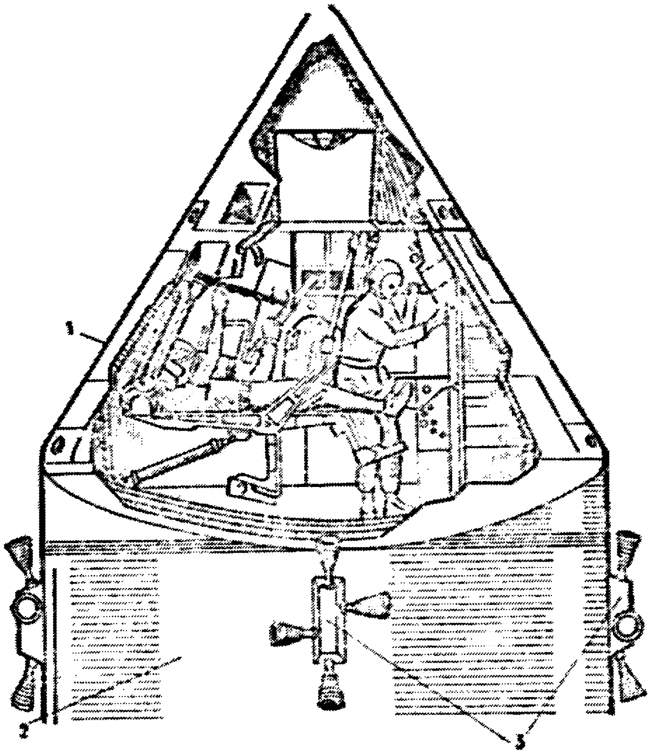

Figure 21. Arrangement of the Auxiliary Engines of the "Apollo" Spacecraft Service Module Jet Control System: 1 -- crew module; 2 -- service module; 3 -- auxiliary liquid rocket engines

operations. The engines of the control system also provide a specific increase in the speed of the takeoff stabe in the event of premature shutdown of the main engine.

The R-4D engine operates on aerozine-nitrogen tetroxide or monomethyl hydrazine-nitrogen tetroxide fuel and develops a thrust of 45 kg at a chamber pressure of approximately 7 atm. The engine weighs 2.3 kg, its height is 34 cm and its diameter is 16.5 cm. The engine consists of a chamber and solenoid oxidizer and fuel valves installed on it. There is a total of 70 components in the engine structure.

The output section of the liquid rocket engine jet nozzle is manufactured of stainless steel and the remaining part of the chamber housing is manufactured from molybdenum alloy and is covered on the inside with an antioxidizing ceramic composition (molybdenum disilicide). The chamber is designed for radiant cooling in combination with cooling by protective film of fuel and operates at gas temperature of 1150°C, which is considerably below the temperature corresponding to production of maximum specific impulse. The reduced temperature, which provides chamber reliability, is achieved by selecting a specific ratio of fuel and oxidizer flow rates.

The R-4D engine may function in both the continuous and pulse thrust modes. A total of 400 of these engines, the total operating time of which was

9 hours and the number of firings of which was approximately 750,000, was used in the "Apollo" program. The total operating life exceeded 7 minutes, the pulse repetition rate reached 30 per second with pulse length of 0.01 second and the number of firings was 35,000 for individual models of the engines (at different flight stages).

Strict evaluation of the utilization efficiency of the fuel consumed by the engine during a single operating cycle is required with such a high number of engine firings. Uneconomical fuel utilization during firing and shutting off the engine (i.e., during transient operating modes) may lead to a significant decrease of its specific impulse and consequently to the need to increase the fuel reserves on board the spacecraft and in the final analysis to a decrease in the payload.

The nature of firing and shutting down the engine is determined to a great extent by its design. When designing liquid rocket engines operating in the pulsed mode, one must attempt to achieve a maximum control valve response rate (minimum time) and minimum volume of spaces between valves and the operating zone of the combustion chamber (this is also desirable for other reasons).

The high speed of the R-4D engine control valves (their response time is less than 0.01 s) is provided by the presence of only two moving parts (a rod and spring) of small mass in each of them. The minimum volume of the mentioned spaces is achieved by installing the valves directly in the mixing head (so that it is at the same time part of the valve design) and in the adopted chamber cooling scheme.

However, uneconomical fuel utilization during starting and shutting down the engine could not be completely eliminated (this is impossible with regard to the nature of the operating process itself). The specific impulse of the R-4D when operating in the continuous thrust mode is 2,750 m/s and that when operating in the pulsed mode is 1,960 m/s; the specific impulse during the total operating time (average) is 2,160 m/s.

Microcracks, caused by the nature of its loading, usually appear in the chamber design when the engine is functioning in the pulsed mode. Development of these microcracks limits the engine operating life. When selecting molybdenum disilicide as the coating for the R-4D chamber wall, its capability to "heal" the microcracks was taken into account. This effect is achieved by formation of silicon dioxide in the crack zone, which protects the design material of the chamber wall against oxidation.

The R-4D is an example of a unversal liquid rocket engine. It was used not only in the "Apollo" spacecraft but also in the "Lunar Orbiter" spacecraft (one of the American lunar programs) and also during flights of the "Skylab" program. The R-4D engines installed in gimbal suspensions in the "Lunar Orbiter" performed functions inherent to both auxiliary and main engines.

And now a few words about the SE-8 liquid rocket engine of the Rocketdyne Company. Twelve of these engines are used on the "Apollo" spacecraft in the crew module jet control system. The engine operates on monomethyl hydrazine-nitrogen tetroxide fuel and develops a thrust of 42 kg. The chamber is designed for ablation cooling in combination with fuel-film cooling and operates at approximately the same temperature as the R-4D chamber. A refractory lining on a graphite base is installed in the minimum cross-section of the chamber. The maximum specific impulse of the SE-8 is 2,650 m/s.

The R-4D and SE-8 engines which we discussed are related to the largest auxiliary engines operating on bipropellant. Among the smallest engines may be named those of the West German Messerschmitt-Belkoff-Blum Company, used in the orientation system of the "Symphonia" communications satellite (designed to operate for 5 years). This engine weighing 160 g operates on the same fuel as the SE-8 and develops a thrust of 1 kg at maximum specific impulse of 2,870 m/s.

Liquid Rocket Engines for Astronaut Propulsion

Investigations to develop liquid rocket engines to propel astronauts in open space are being intensively conducted by American companies. The MMU backpack propulsion unit, which operates on hydrogen peroxide, was successfully tested in 1966 during flight of one of the "Gemini" spacecraft. It contained 12 liquid rocket engines of the Walter Kid Company with a thrust of 1 kg each.

Among the latest developments should be mentioned the HHHMU propulsion unit of the Rocket Research Company, which operates on a hydrazine-water mixture. Because of dilution of the fuel by water, the temperature of the gases ejected from the engine is only 260°C and the unit is safe for the astronaut. It consists of two liquid rocket engines having a thrust of 0.45 kg and two having a thrust of 0.9 kg. The entire unit is a small device with a compact fuel tank which the astronaut holds in his hand. The fuel tank is easily replaced with a new one; this operation may be performed outside the spacecraft.

Prospects for Development of Liquid Rocket Space Engines

An important role is being allocated in plans for further conquest of space by liquid rocket engines. Powerful liquid rocket engines designed for economic utilization of highly efficient fuels are as before at the center of attention of specialists. Proof of this is the SSME oxygen-hydrogen engine with thrust of more than 200 t, designed to transport the "Space Shuttle" spacecraft and developed by the Rocketdyne Company. This engine, unlike the previous oxygen-hydrogen engines, should function over the entire spacecraft acceleration leg: from start to near-earth orbit insertion. The chamber pressure of the SSME engine exceeds 200 atm. The engine is now in the bench development stage.

Development of cryogenic technology along with advances in the field of heat-insulating materials will soon made development of main and auxiliary engines, operating on cryogenic fuels and developing a high specific impulse,

feasible for spacecraft. A mixture of liquid hydrogen with solid, so-called slush-like hydrogen is of great interest. Rocket fuel density increases when liquid hydrogen is replaced by solid hydrogen and the hydrogen losses to evaporation are reduced several times.

A lot of attention is also being devoted to development of liquid rocket engines designed for use of new more efficient fuels. Significant progress has now been achieved in development of liquid rocket engines with a thrust of several tons, which operate on fuels containing fluorine -- the strongest of known oxidizers. Fluorine-hydrogen fuel exceeds oxygen-hydrogen fuel by approximately 5 percent in specific impulse and double in density. The use of fluorine fuels in which the oxidizer is fluorine, fluorine monoxide or mechanical mixtures of fluorine and oxygen (fluxes) and the fuel is hydrazine, ammonia, diborane or light hydrocarbons (for example, methane), may produce a significant effect in some cases. These fuels provide a specific impulse approximately 10 percent less than oxygen-hydrogen fuel, but their density is just as high as oxygen-kerosene fuel; moreover, they store better under space flight conditions than hydrogen-containing fuels.

However, the development of fluorine fuels requires solution of numerous problems related to the chemical nature of fluorine. This product is exceptionally aggressive. It reacts with almost all organic and inorganic substances with release of high amounts of heat, which frequently causes combustion. Under specific conditions fluorine even reacts with inert gases. Many metals react with fluorine even at room temperature and with insignificant heating (fluorine burns all metals with strong heating in the atmosphere). Asbestos, glass, sand and concrete burn in a fluorine jet; a fire begun as a result of fluorine action is very difficult to extinguish. Fluorine oxidizers and combustion by-products of fluorine fuels are unfortunately related to the most toxic products dangerous to man and the environment.

The chemical aggressiveness of fluorine considerably restricts the number of possible construction materials for liquid rocket engines designed to use fluorine fuels. The inner surfaces of metal structural components coming into contact with fluorine should be subjected to passivation operation, which includes treatment of the surfaces with gaseous fluorine to apply a protective fluoride film. The inner surfaces of the components should have no pores, microcracks, burrs and other defects, since they may cause ignition of the structure. The use of fluorine fuels in liquid rocket engines complicates the problem of developing a reliably cooled chamber since extremely high temperatures are developed during combustion of these fuels (for example, it is 700°C higher for fluorine-ammonia fuel than for oxygen-kerosene fuel and reaches 4100°C).

At the same time, despite the difficulty of developing fluorine fuels, the advances achieved in this field during the past decade provide the basis to assume that fluorine liquid rocket engines will find application during the next few years in the upper stages of rockets designed to transfer automatic

spacecraft to other orbits and to boost them toward the planets. Fluorine liquid rocket engines are regarded as promising engines for spacecraft as well which will be capable of making long flights to the planets.

Ozone, which is also a stronger oxidizer than oxygen, is being considered among the possible fuel components for future liquid rocket engines. Ozone in combination with oxygen provides a theoretically higher specific impulse than the fluorine-hydrogen fuel pair. However, ozone is an extremely explosive product which tends strongly toward spontaneous detonation and, consequently, the problem of production and use of ozone in large quantities must still be resolved.

Improving the characteristics of bipropellant liquid fuels may also be achieved by adding lightweight metals to them as a third component. Of greatest interest among these metal-containing fuels are fluorine-hydrogen-lithium and oxygen-hydrogen-beryllium compositions, which essentially provide production of a specific impulse of approximately 5,000 m/s, close to the maximum for existing molecular fuels. Such a high specific impulse may be explained simply by the large amount of heat released during combustion of metals in oxygen and fluorine and by the low molecular weight of oxygen which receives the dissipated heat.

Numerous problems which include development of appropriate methods of fuel production and storage, organization of fuel feed to the engine chamber and provision of total fuel combustion in the chamber with subsequent efficient dispersion of the combustion by-products, must be solved to develop economical engines operating on metal-containing fuel. A significant disadvantage of the two mentioned metal-containing fuels is their low density, caused by the high hydrogen content (they are four times lighter than oxygen-kerosene fuel).

Along with development of new rocket fuels, a search is being conducted for engineering principles which provide further improvement of liquid rocket engines in economy, overall dimensions and mass. The possibilities of the schemes and design solutions adopted in modern liquid rocket engines are limited in this regard. The fact is that an advantage in the specific impulse and overall dimensions of liquid rocket engines, achieved by increasing the chamber pressure, becomes even less discernible as pressure increases and the difficulties of developing liquid rocket engines increase more and more. A significant increase of chamber pressure above 200-250 atm has little effect and is difficult to achieve.

Liquid rocket engines with external expansion nozzle (Figure 22) are of great interest in this regard. These engines are indebted to their name by the fact that the gas in them flows past the jet nozzle on the outside of the chamber rather than inside the nozzle as in ordinary liquid rocket engines. An exterior expansion nozzle is a shaped body which constricts the gas flow in direction, similar to a tapered or prism-shaped dish with a bottom. The combustion chamber in Figure 22 is in the form of a ring encompassing the nozzle. All the other components of the engine structure, including the

turbopump assembly, are located inside the nozzle. The spent turbine gas is ejected to the outside through openings in the bottom of the nozzle. When the rocket starts, the jet stream is initially pressed against the nozzle by atmospheric pressure and then expands to the sides as the life of the rocket increases. Since the gas flow past the nozzle expands to pressure close to surrounding pressure, the nozzle operates constantly during flight of the rocket in a mode corresponding to the maximum specific impulse, which is a significant advantage of an exterior expansion nozzle compared to those ordinarily used. Another advantage of an exterior expansion nozzle is their significantly smaller overall dimensions (they are three-four times shorter than ordinary nozzles), which is explained by their gas-dynamic characteristics. The use of exterior expansion nozzles permits a significant increase of specific impulse and reduction of overall dimensions of liquid rocket engines without resorting to an increase of chamber pressure above 100 atm. Experimental models of liquid rocket engines with exterior expansion nozzle, designed for thrust of approximately 10 to 100 t, have now been tested. It should be said that development of these engines presents many difficulties to designers and technologists.

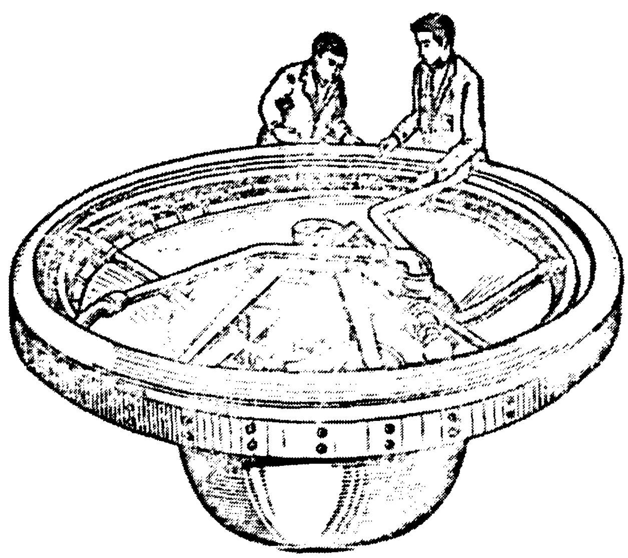

Figure 22. Mock-Up of Liquid Rocket Engine With Exterior Expansion Nozzle (United States)

The development of liquid rocket space engines is related to enormous material expenditures. The cost of developing these engines reaches many hundred million dollars and the cost of serially-produced models is frequently expressed by seven-figure numbers. Nevertheless such expensive articles together with other rocket and spacecraft components are used only once. Investigations of projects for multi-use space transport systems, carried out during the past few years, showed that their development is justified economically only if they are used frequently. Investigations of multi-use space systems recently entered the stage of developing the first models, use of which is planned for the early 1980's. One of the key problems in

developing these systems is to develop powerful efficient liquid rocket engines designed for several tens of flights and for an operating life of several hours with a small number of adjustments between flights. Development of a special system for technical diagnosis of the status of liquid rocket engines becomes necessary under these conditions. A new element appears in the engine: a control unit with a small computer which controls engine operation and issues instructions for emergency shutdown if necessary.

The first models of multi-use space systems, like existing spacecraft, will be multistage. Some of their main components as before are designed for one-time use. An example is the American "Space Shuttle" space transport system. Multi-use space systems which do not contain jettisonable parts and which completely justify their name,* will probably be developed by the end of the century.

The propulsion plants of the systems indicated above and systems as a whole, whose operation brings a significant economic advantage, are in themselves expensive facilities which embody the latest advances of science and technology. Along with this development of ordinary expendable rockets, imperfect in the engineering sense but not requiring large expenditures of funds and time for development, may be feasible for future space programs. An experimental model of an inexpensive liquid rocket engine for a rocket with thrust of 113 t, designed for pressure feed of fuel at a chamber pressure of 21 atm, was tested on a test stand in the United States in 1968; this engine is similar in design to the landing engine of the "Apollo" lunar module. The cost of the engine was a little more than 20,000 dollars. Let us point out for comparison that the serially-produced J-2 engine with thrust of 104 t cost more than 1 million dollars.

Thus, the possibilities of developing liquid rocket space engines are far from exhausted. It should be taken into account that we have considered mainly only those prospects for development which are related to investigations already begun.

In conclusion let us say a few words about future liquid rocket space engines in general, taking into account the circumstance that other types of engines capable of being used in cosmonautics exist and are being developed.

Conclusions

Liquid rocket space engines appeared about 50 years after the idea of space flight had been scientifically and technically advanced. Numerous models of liquid rocket space engines, differing significantly between each other in external appearance, design and characteristics, were developed after launch of the first artificial earth satellite. Along with engines which developed a thrust of a fraction of a gram and which will fit in the palm, there are

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*For more details see V. I. Levantovskiy's pamphlet "Transportnyye kosmicheskiye sistemy" [Space Transport Systems], Moscow, Znaniye, 1976.

engines with a thrust of 100 tons and height of several meters which weigh many tons. The operating life of low-thrust liquid rocket engines reaches several hours and the number of firings reaches many thousand, whereas powerful liquid rocket engines are usually fired once and operate less than 10 minutes.

Compared to RDTT, also used extensively in cosmonautics, liquid rocket engines have the advantage that they develop a higher specific impulse and can operate in their most diverse modes with multiple firings. The advantages of RDTT are determined by their design simplicity, ease of storage in a charged state and the high density of solid fuel.

Liquid rocket engines and RDTT are related to chemical rocket engines, since they develop thrust by using the potential chemical energy of the fuel. This energy reserve restricts the specific impulse of the engine to approximately 5 km/s, which is considerably less than orbital velocity. In this regard enormous and complex rockets whose payload is some fraction of the launch mass, must be constructed for space flight.

A significant increase of the specific impulse of chemical rocket engines (up to 10-20 km/s) may be expected only if hypothetical fuels are created which contain free atoms and radicals or excited atoms and molecules. However, real types of rocket engines with high specific impulse are known. We are talking about nuclear and electric rocket engines.

The modern level of science and technology permits development of nuclear rocket engines (YaRD) with a thrust of several tens of tons and specific impulse of approximately 8 km/s. The working substance which develops thrust in these engines is hydrogen, which is stored in the liquid state in a tank. Upon passing through a nuclear reactor, the hydrogen is transformed to a high-temperature gas which is then accelerated in a jet nozzle. A nuclear rocket engine may be used as the main engines of spacecraft and the upper stages of space rockets. One of the most developed projects for using nuclear rocket engines provides for expeditions to Mars.

Along with the working substance, electric energy used to heat or boost the working substance, is employed in electric rocket engines (ERD) to create thrust. Because of the low thrust to weight ratio, ERD may be used only in spacecraft. Numerous models of electric rocket engines are now being used as auxiliary space engines. They develop a thrust up to several kilograms and specific impulse up to several tens of kilometers per second.

Investigations showed that electric rocket engines may also be used as the main engines of spacecraft. With regard to the low acceleration imparted to the spacecraft, these engines should operate continuously for a long time: for example, more than 1 year when launching a satellite to Jupiter.

Rational combination of nuclear and electric rocket engines with modern chemical engines will provide further progress in cosmonautics.

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